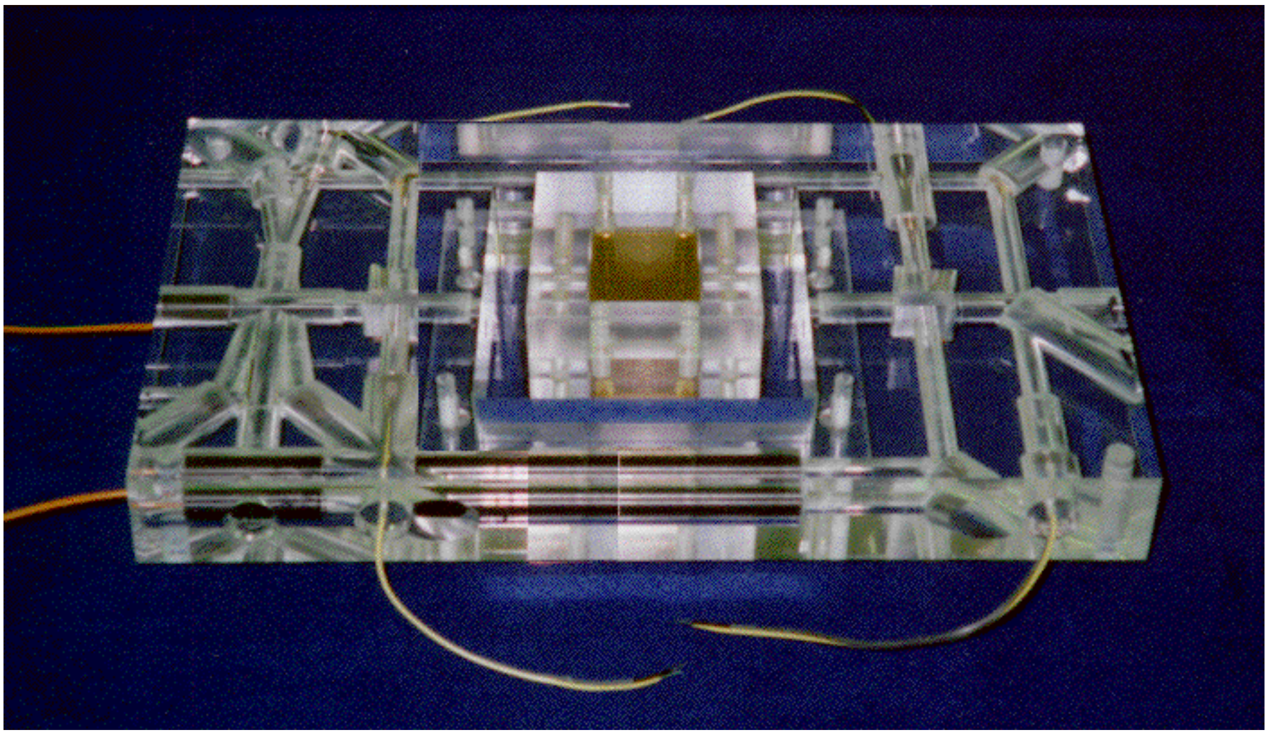
LISA

Laser Interferometer Space Antenna

Technology Plan



February 4, 1999

Jet Propulsion Laboratory
California Institute of Technology

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Technology Plan

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Executive Summary

The Laser Interferometer Space Antenna (LISA) is designed to detect and study low-frequency gravitational radiation. The types of exciting astrophysical sources potentially visible to LISA include extra-galactic massive black hole binaries at cosmological distances, binary systems composed of a compact star and a massive black hole, galactic neutron star-black hole binaries, and background radiation from the Big Bang. LISA will also observe galactic binary systems which are known to exist.

LISA will complement ground-based gravitational-wave observatories such as the Laser Interferometer Gravitational-wave Observatory (LIGO) and several other currently under construction and planned for operation about the year 2000. The major ground-based detectors under construction are based on using laser interferometry to measure changes in the distances between isolated proof masses. Because it is impossible to isolate proof masses from the slowly varying gravitational potential of the Earth, it will be impossible for ground-based detectors to observe low-frequency gravitational waves (frequencies below ~1 Hz). Searches for gravitational waves with frequencies from 0.001 Hz to 1 Hz have been previously made using Doppler measurements to interplanetary spacecraft. The Cassini project will include a special radio system to achieve a one order of magnitude improvement in gravitational-wave sensitivity compared with previous measurements. LISA will have six orders of magnitude greater sensitivity than the Cassini gravitational-wave experiment.

The LISA project has been under study since 1994 by a team of European and US scientists. LISA was selected by the European Space Agency as a future Cornerstone mission. LISA is also included in NASA's strategic plan as a future mission within the Structure and Evolution of the Universe program. A shared NASA/ESA mission is envisioned though formal negotiations have not yet taken place.

The LISA science goals depend on measuring changes in the distances between proof masses separated by millions of kilometers with picometer accuracy. These proof masses must be extremely well isolated from non-gravitational disturbances. Achieving these goals requires extending the current state of the art in precision measurements by several orders of magnitude.

The technology needed for LISA is based on extrapolations of techniques developed for ground-based gravitational-wave observatories, and on technology used in spacecraft Doppler tracking and other precision spacecraft measurements. However the extrapolations are over several orders of magnitude. While construction of the LISA project could, in principle, begin immediately, there would be significant risk associated with the technology extrapolations.

The risk associated with the LISA technology can be significantly reduced through a targeted program of ground and space technology development and validation. Neither NASA nor ESA has yet instituted a technology development program specifically for LISA, though both agencies have general technology programs which include developments that are directly applicable.

The LISA Technology Plan specifies a range of technology development and tests to reduce the risk associated with the project as much as possible. Developments already funded within the US or Europe are noted. Without a negotiated sharing between NASA and ESA, there is no basis for

allocating the costs for the needed development. Costs for developments not yet funded are estimated based on implementation by NASA. It is expected that some of the developments will be funded by European agencies.

The Technology Plan is based on an assumed New Start in 2006 for launch in 2008. The plan also assumes that there will be an opportunity to perform flight demonstrations of some of the key technologies. The full cost of such a flight demonstration is beyond the scope of the plan. Such a flight demonstration is being considered for the fifth in the series of Deep Space projects under NASA's New Millennium program, and is also being pursued within Europe.

The Technology Plan was prepared by the LISA Pre-Project office at JPL. It is based on the efforts of the European LISA Science Study Team, the US LISA Mission Definition Team, supported by personnel from a large number of institutions.

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1. Introduction

1.1 Purpose

The purpose of the Laser Interferometer Space Antenna (LISA) Technology Plan is to document the mission technology needs, the current state of technology development, and the work that needs to be done to reach the required technology readiness level prior to start of spacecraft construction (project phase C/D). The LISA Technology Plan describes the technology development objectives, the development approach, and summary implementation plans for the various technology areas.

The LISA mission is foreseen here as an international partnership primarily between NASA and ESA, with participation from European national space agencies. There may also be contributions from other nations. Relevant technology development is underway in the US and Europe. The LISA Technology Plan describes the technology status and ongoing efforts on both continents. The further development necessary development is outlined without stating where the development will take place. To date LISA technology development has been coordinated informally by the US and European LISA science teams. Future technology development will depend on funding by the various agencies and will need to be coordinated by negotiated agreements between agencies.

1.2 Applicable Documents

The current LISA mission and spacecraft design are described in the LISA Mission Concept Study. The science goals and instrumentation are more fully developed in the LISA Pre-Phase A Report. The instrument and interfaces are described in more detail in the LISA Payload Definition Document.

1.3 LISA Mission Description

1.3.1 Mission Objectives

The major science objectives of LISA are to observe low-frequency gravitational waves from;

a) Galactic binary systems including black holes, neutron stars, and interacting white dwarfs that are possible progenitors of Type I supernovae.

b) Mergers of massive black holes (100-10^7 $M_\odot$) such as found in the centers of many galaxies

c) Interaction of 1-10$M_\odot$ black holes and massive black holes, providing precision measurements of strong-field gravitation

d) Formation of massive black holes, for which there are several candidate mechanisms

These sources produce gravitational radiation in the frequency band 10^{-4} Hz to 10^{-1} Hz which cannot be observed by ground-based detectors since the variation in local gravity at these frequencies is larger than the expected signals.

Gravitational wave are propagating gravitational fields, "ripples" in the curvature of space-time which cause a variable strain of space-time, which result in changes in the distance between points, with the size of the changes proportional to the distance between the points. LISA will detect the changes in distance between proof masses separated by 5 million kilometers. The distance between the proof masses will be measured using laser interferometry. The nominal sensitivity for LISA is shown in Figure 1.1 below. At the lowest frequencies the LISA sensitivity will be limited by displacement of the proof masses by noise forces such as fluctuations in the solar radiation pressure. For frequencies above 2×10^{-3} Hz the sensitivity will be limited by noise in the interferometer measurements, including the laser shot noise. For frequencies above 3×10^{-2} Hz the sensitivity is reduced because at the higher frequencies the wavelength of the gravitational radiation is shorter than the distance between proof masses.

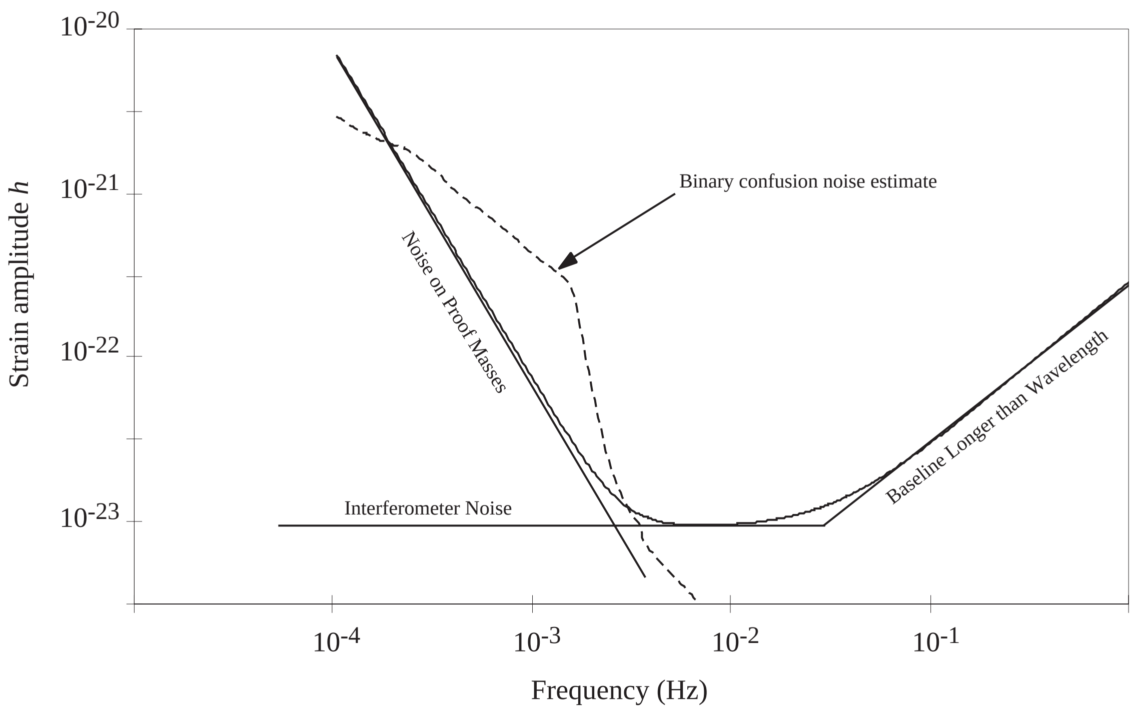

Figure 1.1 LISA sensitivity for gravitational waves of strain amplitude *h* with signal-to-noise ratio of 5 after one year of observation.

The interferometer noise limit is based on the assumption of 1 W laser and 30 cm diameter telescopes for transmission and reception of laser signal between the proof masses. The proof-mass noise is based on models of expected noise forces. The low-frequency sensitivity based on noise forces can be reduced by increasing the distance between proof masses, at the cost of reduced sensitivity at the higher frequencies. The choice of 5 million kilometers for the separation between proof masses has been made as a compromise between increased science return at low frequencies and cost and technical issues associated with larger separations.

1.3.2 Reference Mission Design

The LISA mission will comprise three spacecraft located 5×10^6 km apart forming an triangular-shaped Michelson interferometer (Figure 1.2). The spacecraft orbits are selected such that the triangular formation is maintained throughout the year with the triangle appearing to rotate about the center of the formation once per year. The center of the triangle formation will be in the ecliptic plane 1 AU (150×10^6 km) from the Sun and 20° behind (52×10^6 km) the Earth.

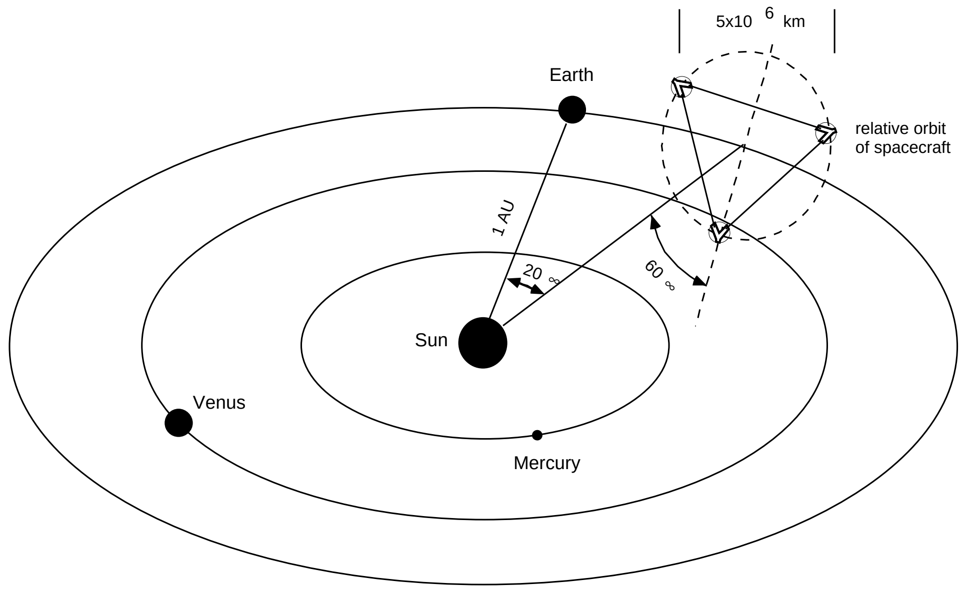

Figure 1.2 Schematic diagram of the LISA configuration. Three spacecraft form an equilateral triangle with sides 5 million km in length. The plane of the triangle is tilted by 60° out of the ecliptic. The two optical assemblies on one spacecraft combine with an optical assembly from each of the other two spacecraft to form a Michelson interferometer. The drawing is not to scale.

Figure 1.3 shows the LISA spacecraft. Each spacecraft will contain two optical assemblies, each of which in turn will house a proof mass, centered in an optical bench, and a 30-cm diameter telescope. The relative displacements between the spacecraft and the proof masses contained within it will be measured electrostatically. Micronewton thrusters will be used to keep the spacecraft centered on the average position of the two proof masses. This ‘drag-free’ operation is necessary to keep non-gravitational forces on the proof masses at an acceptable level. The nominal thrust level is about 20 micronewtons to cancel the force due to solar radiation pressure. Less than one kilogram of propellant will be needed during the science mission.

The three LISA spacecraft are to be launched on a single Delta-II 7925H. The initial orbit will have an excess energy of $C_3 = 1.1\ km^2/s^2$ so that the three spacecraft will slowly drift behind the Earth. At launch, each spacecraft will be attached to a propulsion module, with each spacecraft/propulsion module combination separating from each other after injection into the transfer orbit. The propulsion modules provide the capability to maneuver the spacecraft into the final orbits. After reaching the final orbits, about 13 months after launch, the propulsion modules will be separated from the spacecraft. During the 3 year prime science mission, the spacecraft orbits will evolve under gravitational forces only.

The distances between proof masses in different spacecraft is measured using laser interferometry. Each spacecraft can act as the vertex of a Michelson interferometer with ends defined by a single optical assembly on each of the other two spacecraft. Of the three possible interferometers, two will be independent, giving information about both polarizations of received gravitational waves.

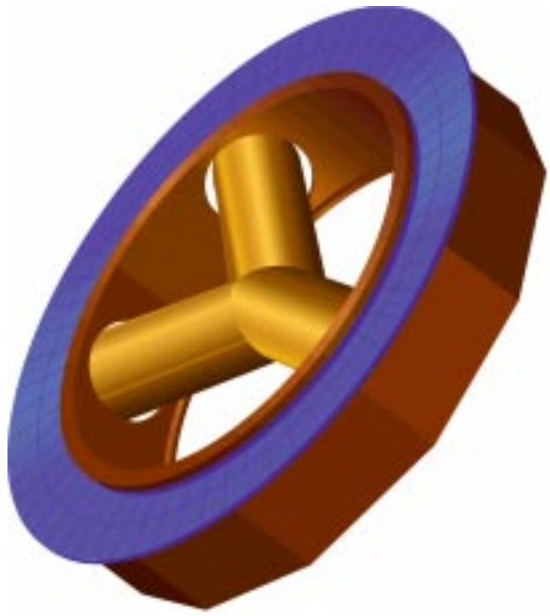

Figure 1.3 Artist's concept of the LISA spacecraft. Not shown is a cover over the top of the cylinder that prevents sunlight from striking the Y-shaped payload enclosure.

Data on the measured distance between the test masses are continuously acquired throughout the mission. Pre-processing of the data is done by the spacecraft computer to remove the laser phase noise and reduce the signal bandwidth. The data are stored in the spacecraft computer memory. The current plan is for the data to be transmitted to Earth every other day. A single 10.5-hour tracking pass of a Deep Space Network (DSN) 34-m antenna would be used to download science and housekeeping data from each spacecraft.

Key Reference Mission Parameters:

- Three spacecraft in triangle formation, separation 5×10^6 km
- Earth-trailing orbit, 1 AU from Sun, 0.3 AU from Earth
- Launch Vehicle Delta II 7925H
- Spacecraft Mass 265 kg (30% contingency)
- Spacecraft Power 200 W (30 % contingency)
- Launch Mass 1400 kg (30% contingency)
- Tracking/Navigation via Deep Space Network 34m antenna
- Lifetime: 1 year cruise, 3 year prime mission, consumables for 7 year extended mission

1.3.3 Reference Instrument Design

The instrument for each spacecraft consists of two optical assemblies, a surrounding structure, and lasers on a radiator separately attached to the spacecraft. Figure 1.4 shows a cross section of the two optical assemblies as they will be mounted within the spacecraft. Figure 1.5 provides more information about the optical assembly. Each optical assembly contains a 30 cm diameter f/1 Cassegrain telescope. Each optical assembly also has an optical bench, machined from a block of Ultra-Low Expansion (ULE) glass with dimensions $20 \times 35 \times 4$ cm, which contains injection, detection and beam-shaping optics. A inertial sensor (or "accelerometer") is mounted to the center of each optical bench. The proof mass of the inertial sensor acts as the mirror at the end of the interferometer arm.

The laser beam is carried to the optical bench within each optical assembly by an optical fiber. A few mW is split off the 1 W main beam to serve as the local reference for the heterodyne measurement of the phase of the incoming beam from the far spacecraft. Also, a few mW is split off and directed towards a triangular cavity which is used as a frequency reference. The incoming light from the telescope is reflected off the proof mass and superimposed with the local laser on the phase-measuring diode. A small fraction (a few mW) of the laser light is reflected off the back of the proof mass and sent for phase-comparison with the other optical assembly via an optical fiber. By bouncing the laser beams off the proof mass in the manner described, the interferometric measurement of proof mass position is, to first order, unaffected by motion of the surrounding spacecraft.

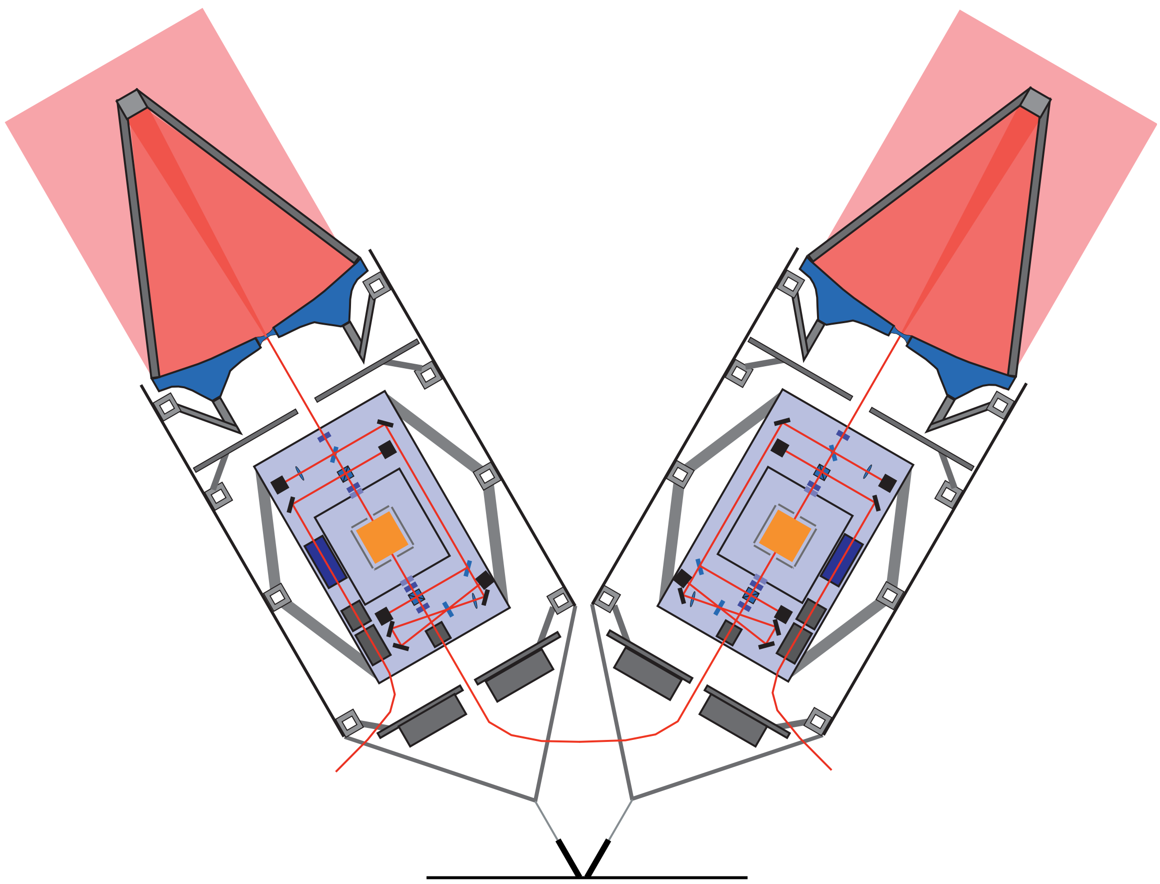

Figure 1.4 Cross section of the two optical assemblies comprising the main part of the payload on each LISA spacecraft. The two assemblies are mounted from flexures at the back (bottom of figure) and from pointing actuators (not shown) at the front, near the primary mirrors.

The inertial sensors may be derived from accelerometers developed by the Office Nationale de'Etudes et de Recherches Aerospatiales (ONERA). Alternatively the sensors could be derived from gyroscopes developed for Gravity-Probe B, which will be used as inertial sensors for the spacecraft position control. The nominal 4-cm cubic proof mass will be made of a special gold-platinum alloy with magnetic susceptibility less than 10^{-6} to reduce noise forces on the proof mass as it moves through the variable solar magnetic field. The proof masses will freely float inside a

ULE housing, which supports electrodes used to sense the position of the proof mass through changes in the capacitance between electrodes. The ULE housing will be enclosed in a titanium vacuum chamber, which will be vented to space to keep the interior pressure less than 10^{-6} Pa (10^{-8} mbar). Electrostatic charging of the proof mass due to cosmic ray protons with energies in excess of 100 MeV would cause noise as it moves through the solar magnetic field. The proof mass charge will be kept small by directing ultraviolet light from a mercury discharge lamp at the proof mass and walls, similar to the approach developed for Gravity Probe B.

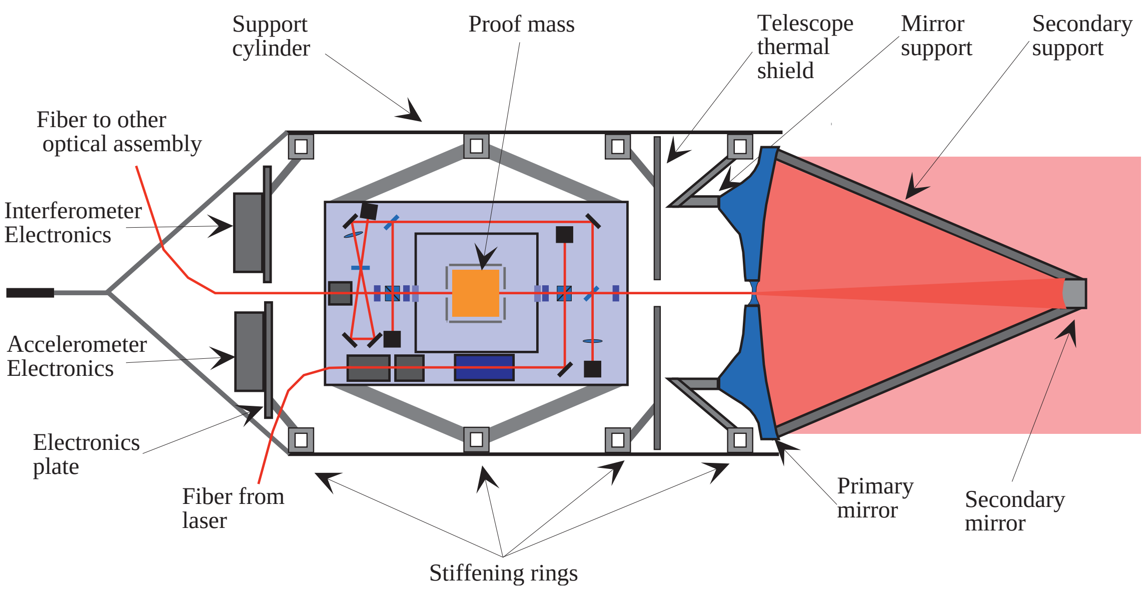

Figure 1.5 Cross section of one optical assembly.

At each optical assembly, heterodyne phase measurements are made to determine the difference in frequency between the local laser and the lasers on other spacecraft. One laser on one optical assembly will serve as the master and will be locked to the an onboard reference cavity. The lasers on the other optical assembly, and on the other spacecraft, will be phase-locked to the master laser via the phase comparison beam exchanged between the incoming beams and the local laser. Linear combinations of the phase measurements are made to determine the difference in the distances between proof masses on different spacecraft, which is done implicitly in a Michelson interferometer. Distance changes will be measured continuously with picometer precision, and will be transmitted to the Earth for analysis.

1.3.4 Major Technical Challenges

The LISA science objectives lead to the measurement requirements, which are to determine the changes of distance between proof masses separated by 5×10^{6} km with picometer precision over a frequency range of 10^{-4} Hz to 10^{-1} Hz. The technology challenges are thus the development of a system to measure changes in distances between proof masses, and of isolating the proof masses from external disturbances so that changes in their separation due to gravitational waves are not masked by motions due to other forces. The technology challenges are summarized in Figure 1.6.

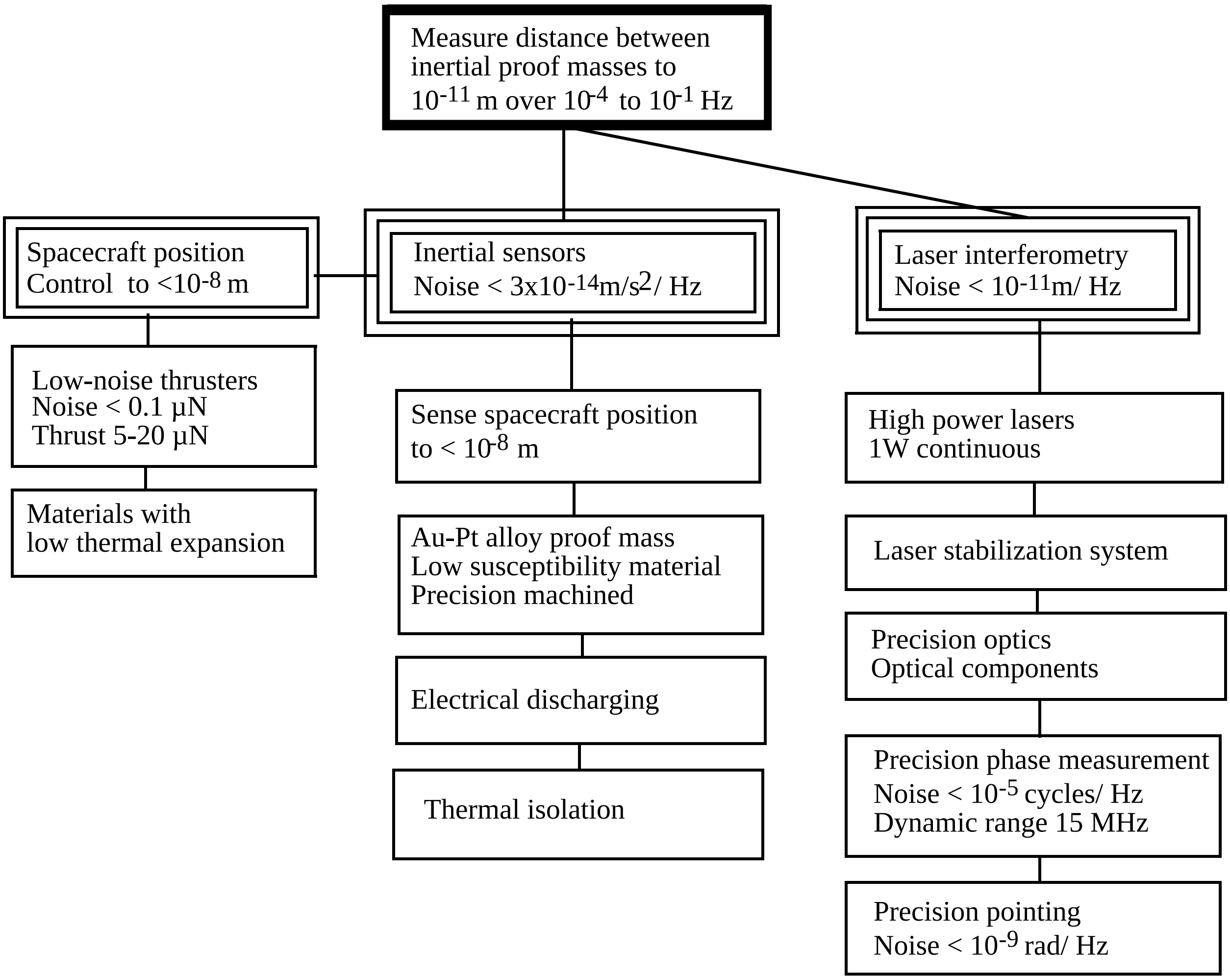

Figure 1.6 Areas of technology development for LISA. The top level requirements for the science mission lead to the measurement requirements on the inertial sensor and interferometry. The inertial sensor performance in turn places requirements on the spacecraft position control, which requires additional technology.

The major LISA technology challenge is the need for proof masses sufficiently isolated from non-gravitational forces. The desire to measure distance changes, due to gravitational waves, of order 10 picometers means that the non-gravitational forces on the proof masses need to produce accelerations less than 3×10^{-14} m/s^2/ Hz at a frequency of 10^{-3} Hz. This level of performance cannot be established in laboratory testing, since extremely small changes in instrument orientation variably couple in the 10 m/s^2 acceleration due to the Earth's gravity. The best laboratory performance achieved for a complete inertial sensor, over the frequency range of interest for LISA, is $\sim10^{-10}$ m/s^2/ Hz. However the types of noise forces that can affect the inertial sensor proof mass can be identified and separately characterized in laboratory tests. Based on the noise models and tests, detailed instrument designs for inertial sensors meeting the LISA requirements have been completed (Rodrigues and Touboul 1998). Because the LISA requirements are far beyond what can be achieved in the laboratory, it is highly desirable to have a space flight experiment to demonstrate that the performance specifications can be met.

Many of the noise forces on the proof mass are affected by fluctuations in the distance between the proof mass and the rest of the spacecraft. For example, the spacecraft mass has a gravitational pull on the proof mass and thus fluctuations in the position of the spacecraft cause fluctuations in the force on the proof mass. Because of this the position of the spacecraft must be controlled to stay centered on the proof mass. The position control requirements, derived from the inertial sensor requirements, in turn place requirements on the spacecraft thrusters. With the current mission design, the thrusters are required to have a thrust noise of about 0.1µN, with a continuous thrust of about 25 µN in order to oppose the force from solar radiation pressure. The best candidate thrusters for meeting these requirements are based on emission of ionized metal (Cs or In) atoms accelerated by an electric field. These thrusters have been under development for many years, originally with the idea of more efficiently performing attitude control for commercial satellites. Versions of the thrusters are undergoing laboratory tests, and several flight experiments are also planned, as described below.

Measurement of changes of distances between proof masses is routinely done with laser interferometry. For ground-based gravitational wave detectors, techniques for measuring distance changes of order 10^{-19} m have been developed and demonstrated over the past 20 years or more. However, interferometers for ground-based gravitational-wave detection have been optimized for motions at much higher frequencies than desired for LISA, and at much higher laser signal powers. The LISA measurement band, and many of the technological challenges, are perhaps more similar to spacecraft Doppler tracking which uses radio signals to measure changes between the Earth and interplanetary spacecraft, which is a technique which has been used to look for low-frequency gravitational waves. Like Doppler tracking, LISA will transmit and receive signals, albeit at optical rather than radio frequencies, between pairs of antennas and use heterodyne mixing of signals to measure Doppler shift. By being outside the Earth’s atmosphere, and by using much higher transmission frequencies, LISA will not be affected by fluctuations in the transmission media (Earth’s troposphere and ionosphere, and solar plasma) that are one of the limiting errors sources for Doppler tracking. And like ground-based interferometers, and unlike Doppler tracking experiments, LISA will use two pairs of transmitter/receiver satellites, synthesizing a two-arm Michelson interferometer, to cancel the noise due to frequency fluctuations of the transmission signal, to achieve the desired sensitivity. There are a number of error sources associated with the low-frequency interferometry to be used for the LISA distance measurements. These noise sources and a set of planned measurements to demonstrate a complete working system, are described below.

1.3.5 Project Schedule and Budget

The LISA project schedule and budget are not yet approved. Within the NASA Strategic Plan, New Start (start of Phase C/D) could not be before 2005. The assumed mission schedule shown in Figure 1.7 assumes a New Start in January 2006, and launch 30 months later in July 2008. Technology readiness is to be established by July 2005, with the Non-Advocate Review (NAR) and Preliminary Design Review (PDR) shortly thereafter.

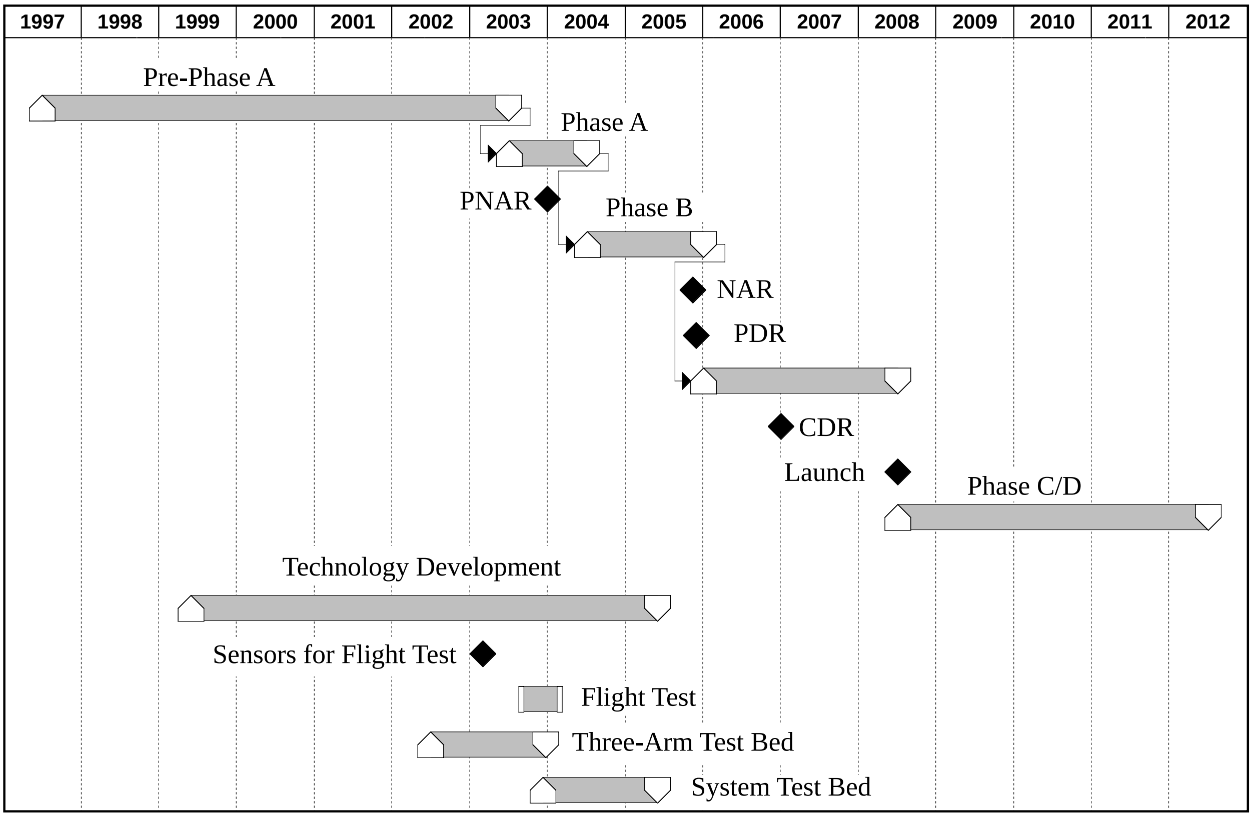

Figure 1.7 Top-level LISA project schedule (calendar years)

The total life-cycle LISA project budget is shown in Table 1.1. This budget is based the LISA Mission Concept Study but with lower telecommunications and tracking cost. The lower costs are based on the assumption that the Mars Pathfinder antenna can be used for the LISA spacecraft, and a 20-W X-band solid-state power amplifier currently under development. These components, if available, will thus not have to be developed by the LISA project. They will also allow for a higher data rate than assumed for the Mission Concept Study, which will lower tracking requirements and costs. Separately shown in Table 1.1 is the budget developed in this Technology Plan for activities not currently funded, and for a Flight Demonstration.

Table 1.1 LISA project budget

									FY 99 $M							
	1998	1999	2000	2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	2011	2012	Total
Pre-Phase A	0.2	0.5	0.9	1.5	2											5.1
Phase A						2.8	8.4									11.2
Phase B							3.7	14.9	3.7							22.3
Phase C/D									73.6	98.2	73.6					245.4
Phase E											1.7	7.0	7.0	7.0	5.8	28.4
Launch Vehicle								3.1	23.5	24.8	8.6					60.0
Reserves						0.14	0.8	1.5	13.6	19.6	14.9	0.7	0.7	0.7	0.6	53.2
Total	0.2	0.5	0.9	1.5	2.0	2.9	12.9	19.5	114.5	142.6	98.8	7.7	7.7	7.7	6.4	425.6
Technology Development		0.3	3.7	7.0	9.5	7.2	4.1	1.5								33.3
Flight Demonstration (DS-5)		0.5	2.0	8.0	8.0	8	2									28.5

The overall LISA budget is dependent on the assumed schedule with launch in 2008. A later launch would result in higher real-year costs due to inflation. The budget for technology development is based on the budget developed in this document. The cost of efforts currently funded within Europe are not included here. It is expected that approximately half of the funding required for the project will come from NASA and half from ESA and its members states, though no detailed sharing arrangements have yet been negotiated. The budget for a technology flight demonstration is shown separately, based on the assumption that the budget allocated for the New Millennium DS-5 mission will allow for a demonstration of the key LISA technologies. This will be studied separately by the New Millennium program in the first half of 1999.

1.4 Relationship to Other Missions

LISA will be a pioneering mission using laser interferometry for space detection of gravitational waves. As such many of the technology goals are specific to this unique mission, though some specific technology can be adapted from other missions.

The TRIAD mission, launched in 1972, is the only three-axis drag-free satellite flown. Many of the ideas for the LISA inertial sensor and spacecraft control are derived from engineering studies done for TRIAD. TRIAD demonstrated a disturbance reduction system based on capacitive sensing of a spherical test mass with long term acceleration noise of 5×10^{-11} m/s^2 or less. That disturbance reduction system was built jointly by Applied Physics Laboratory and Stanford University.

The Space Interferometer Mission (SIM) is developing a great deal of technology related to laser interferometry. SIM is designed to measure the angles between stars with micro-arcsecond resolution. This relies in part on being able to measure the distance between telescope elements separated by ~10 m with an accuracy of 0.1 nm. Some of the challenges of the SIM laser metrology are not relevant for LISA, such as frequent large changes in telescope pointing and spacecraft attitude, and absolute length measurements. Also the SIM lasers operate at lower power and longer wavelength than desired for LISA. However many of the components developed and tested by SIM can be used for LISA as well as the engineering experience in developing interferometers for space.

The Gravity Recovery and Climate Experiment (GRACE) has many similarities to the LISA mission. GRACE involves two spacecraft measuring changes in the distance between them to recover information about the Earth's gravity field. The two GRACE spacecraft will be in low polar orbits, with spacecraft separation of ~400 km. Each spacecraft will transmit a microwave signal with wavelength ~1 cm and receive a similar signal from the other spacecraft. The difference in carrier phase will be measured at each spacecraft and combined to form a one-arm microwave interferometer much like on of the LISA arms. Each spacecraft also carries an accelerometer to measure distance changes induced by atmospheric drag. These accelerometers are similar in design to candidate inertial sensors for LISA. The signal frequencies of interest for GRACE, which are related to the orbital period, are similar to those for LISA. GRACE will provide validation of inertial sensor performance for LISA at a reduced level of performance. Much of the GRACE phase measurement system can be adapted for use by LISA.

A GRACE-Follow-On mission would be a beneficiary of the planned LISA technology development. To achieve more sensitivity to gravitational effects, the two spacecraft ranging system can use laser interferometry, and inertial sensor, and small ion engines for drag-free operation as developed for LISA.

Much of the technology for Gravity-Probe B (GP-B) can be applied to LISA. The GP-B spacecraft will be controlled to follow the position of one of the gyroscope masses, using electrostatic position sensing and helium gas thrusters. This experience can be applied to the LISA inertial sensors and spacecraft position control. GP-B employs precision optical benches, having developed techniques for reliable precision mounting of optical components to quartz optical benches. The experience in mounting techniques and precision assembly are applicable to the LISA instrumentation.

2. Technology Development Approach

The overall approach adopted for the LISA Technology Plan is to demonstrate the performance of all subsystems at the level required to meet the LISA objectives; to develop and test all components that have not been previously tested for operation in space; and demonstrate system-level performance through a simulated instrument and spacecraft operating in a configuration approximating the mission parameters. Obviously the system cannot be fully demonstrated on the ground, and several of the flight subsystems cannot be operated, much less tested, on the ground due to the Earth's environment. The goal of the Technology Plan is to reduce the risk associated with the development of the LISA subsystems and system integration as much as possible prior to start of Phase C/D.

This will be implemented by design, acquisition, and testing of needed components that lack previous flight heritage; development of breadboard electronics to prove the design and define interfaces needed for flight units; and by a series of ground experiments, including a series of test beds leading to an engineering model of the instrumentation for one spacecraft, to demonstrate performance of individual subsystems and system-level performance. Components whose lifetime is determined to be at risk will be identified, developed and tested for long duration.

The ground environment is most severe for the inertial sensors, since operation of flight units will not be possible in Earth gravity. A space flight demonstration of the inertial sensors is desirable to assure that their performance reaches the expected levels. At time of this writing, this flight demonstration is assumed to take place in association with the New Millennium Deep-Space 5 mission opportunity, although DS-5 selection is yet to take place. If an inertial sensor demonstration is not included as part of DS-5, either another flight opportunity will be needed or else a far more extensive series of ground tests will be needed.

2.1 Inertial Sensor

The desired inertial sensor performance, with accelerations due to noise forces less than $3x10^{-15}$ m/s^2/ Hz in the frequency range 10^{-4} Hz to 3×10^{-3} Hz, cannot be demonstrated on the ground. Sensors have been developed and tested on the ground at the level of 10^{-9} m/s^2/ Hz, which is the practical limit to which the acceleration of 10m/s^2 from the Earth's gravity can be removed by keeping one axis of the sensor orthogonal to the local gravity vector.

Some space testing of inertial sensors has been done, and several missions will be launched in the 2000-2002 time frame that will employ sensors similar to the LISA inertial sensors. These include Gravity-Probe B, whose gyroscope assembly will be also used as an inertial sensor for drag-free spacecraft control, and the CHAMP and GRACE missions which will employ a sensor developed by ONERA to measure the atmospheric drag force.

From experience developed from building these devices, and from specific tests done to measure each possible noise source, it is possible to design an inertial sensor that will meet the LISA requirements. However this sensor cannot be operated, much less tested, in the Earth's gravity, making this a high-risk technology.

It is thus highly desirable to perform a flight demonstration of candidate inertial sensors. Such a test would consist of two inertial sensors, each with a proof mass with its own housing and electronics, on a single spacecraft. The performance would be validated by using one for spacecraft drag-free control, and measuring the position of the second with respect to the first to show that both are following the same trajectory, under the influence of gravitational forces only. Any non-gravitational force would appear as a change in the position of one proof mass with respect to the other.

The LISA technology plan is based on the assumption that a flight test will be performed. The schedule and budget for the inertial sensor technology development is based on the development of two inertial sensors which will be ready for flight on a DS-5 mission in late 2003. The cost of the flight of the two inertial sensors is not considered here.

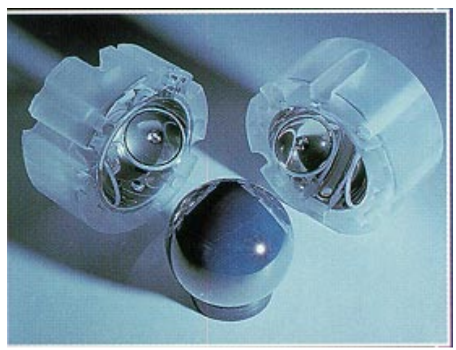

Figure 2.1 Inertial sensor/gyroscope assembly for GP-B

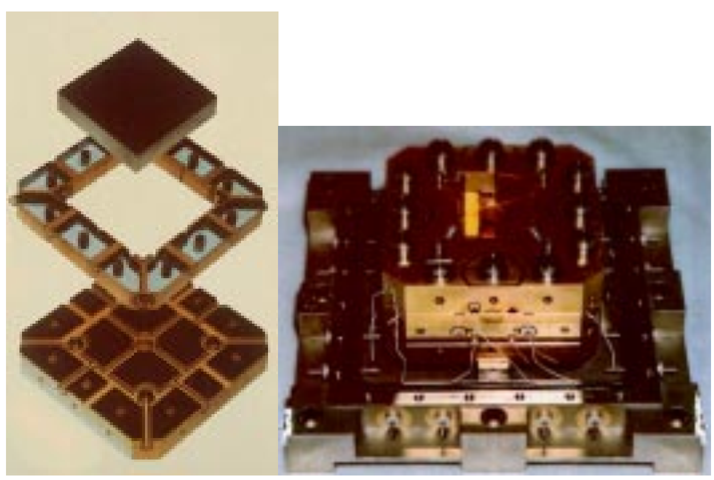

Figure 2.2 GRADIO accelerometer from ONERA.

2.2 Micronewton Thrusters

To achieve the required inertial sensor performance, the position of the spacecraft must be precisely controlled with respect to the inertial sensor proof mass, to reduce noise forces caused by motion of the spacecraft. The spacecraft control in turn requires thrusters with the capability of balancing the solar radiation pressure, with its small variations, with low enough noise such that the spacecraft can be kept centered on the proof masses to the required accuracy (~10 nm/ Hz). The thrusters must also provide very fine spacecraft pointing control. This gives rise to the thruster requirements for thrust controllable in the range 5 μN to 25 μN with noise less than 0.1 μN/ Hz.

The best candidates for meeting these requirements are small ion thrusters which have been developed, partly under funding by ESA, for satellite station keeping and for satellite charge control. Photographs of versions the two candidate devices are shown in Figure 2.2. Both of these devices operate by ionizing and accelerating atoms from a liquid-metal reservoir. These thrusters provide thrust in the desired range, with very high efficiency, allowing a very small amount of propellant to last for more than the 3 year LISA prime science mission.

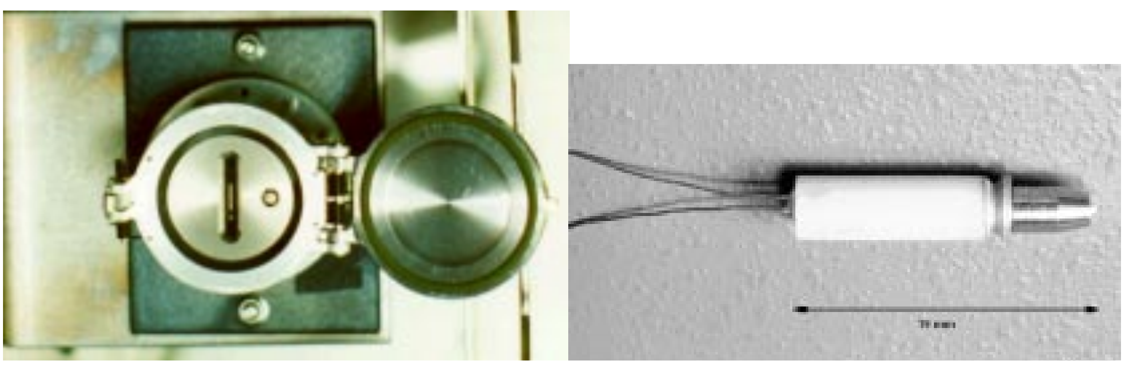

Figure 2.3. Micronewton thrusters developed by Centrospazio (left) and Seibersdorf (right)

The main technology challenges with these thrusters are the controllability of the thrust and their lifetime.

The control issue arise because the thrust, and the thrust noise, has so far not been directly measured. Instead the thrust characteristics have been derived by measurements of the ion current and applied voltage to compute the applied thrust. There are possibly errors in this computation if there are ejecta other than single ionized atoms (i.e. droplets). The thrust is difficult to measure directly because it is so small. However work is underway in Europe to develop torsion balance experiments to directly measure the thrust.

The lifetime issues arise from the high voltages used to ionize and accelerate the atoms to high velocity. There can be erosion of the accelerating grids and ejecting tips. Also the ion stream needs to be combined with an ejected stream of electrons to keep the spacecraft electrically neutral. The device to do this, the neutralizer, had been subject to lifetime limiting effects for larger ion engines and may be a factor here as well.

The purpose of the thruster technology plan is to perform a coordinated series of tests and analysis to ensure that the thrusters meet the LISA mission requirements with a minimum of risk. This plan is develped to meet concerns based on NASA experience with ion engines. Some of those concerns will be addressed by an ESA/ESTEC testing program and will not be duplicated in the US. Some concerns are felt to be more important to NASA engineers than they appear to ESA engineers. Since these are not yet included within the ESA test program, they are here planned to take place within the US. Specific tests to evaluate the performance and lifetime will be performed. It is anticipated that work will be executed at JPL in collaboration with ESA/ESTEC with a minimal duplication of effort.

2.3 Picometer Interferometry

Successful execution of the LISA mission relies on the ability to measure changes in the separation between proof masses to <10 pm/ Hz, down to 10^{-3} Hz. For obvious practical reasons, much of the technology development and validation for LISA will be carried out on the ground, in an environment that is significantly different from the one in which the instrument will be required to perform. Therefore, LISA interferometry development will rely heavily on emulating in-orbit conditions in a laboratory environment. The common paradigm is to use a succession of test beds. While no single test bed can fully simulate the space environment, the overall test program ensures that all critical aspects are covered.

The ground-based laboratory environment makes testing difficult mainly because:

- the presence of air limits the interferometric resolution to ~1 nm, for an optical path of a few meters. A similar limitation applies to laser-gyro measurements, because of airflow along the optical path, which causes non-reciprocal phase shifts.
- typical vibration levels are a few micrometers at a quiet location, with most of the energy concentrated at and around the so called micro-seismic peak, at approximately 0.1 Hz. Practical vibration isolation systems can attenuate vibration transmitted from the ground above ~1 Hz.
- temperature variations in a typical lab are of the order of 0.1^{o}-1^{o} per hour, while LISA-grade performance requires much better thermal stability over several hours.

Also, the separation between components will obviously not be the same as called for in the actual mission.

To develop and validate the technology for LISA, a sequence of tests will be performed in various experimental setups and test beds, as outlined in the table below. Some aspects of the mission may not be testable on the ground. In these cases, one would attempt to develop appropriate models and validate them as well as possible on the test beds.

The scale of the required effort can be assessed by comparison to the technology development plan for the SIM astrometric interferometer mission. In particular, the laser metrology for SIM is comparable in sensitivity and complexity to the LISA interferometer system.

Table 2.1 Interferometry validation approach. Once the required performance has been demonstrated down to 1 mHz for all sensors and for the laser, further subsystem testing is carried out in test beds to the required resolution but only down to 10 Hz, in order to verify that a) integration does not degrade performance and b) control systems perform as desired.

Subsystem	Setup or test bed	Vacuum	Vibration Isolation	1 mK/h Insulation	Comments
Laser stability; 10 Hz/ Hz noise at 1 mHz	2 lasers, 2 reference cavities in vacuum	Y	Y	Y	Rigid cavities insensitive to low-f vibration
Laser phase-locking; 10^{-5} radian / Hz with 50 pW, at 1 mHz	2 lasers	Y	Y	Y	Rigid cavities insensitive to low-f vibration
Measurement accuracy; picometer stability at 1 mHz	Rigid Interferometer	Y	Y	Y	Rigid platform renders low-f vibration common mode
Pointing : 10 nanoradian/ Hz noise; 10 nanoradian control; signal acquisition;	Single-arm test bed	Y	Y	N	Optics integration and control systems test
Interferometer system: 10 pm/ Hz noise at 1 mHz	Three-arm test bed	Y	Y	N	Test functionality of 3-corner interferometer
Control software	Three-arm test bed	Y	Y	N	
Payload system operation; Full-scale payload; Simulated spacecraft; 10 pm/ Hz @ 1 mHz	System test bed	Y	Y	Y	At one corner of 3-arm interferometer

2.4 System Operation

The LISA instrument places continual demands on the spacecraft in terms of position and attitude control. The requirements are extremely tight compared with traditional space missions, yet are fairly easily measured by the inertial sensor and the interferometer, and the micronewton thrusters provide suitable means for the fine control. There are enough separate degrees of freedom and corresponding measurement and control systems to require a detailed modeling and analysis of the system as a whole.

A two-step approach is planned to develop and test the software and control mechanisms of the LISA instrument and spacecraft at the system level. These steps consist of the development of a spacecraft simulator and construction of an engineering model of a complete LISA instrument package. This will allow tests of the operation of the entire instrument system, and of the system interactions of the instrument and the spacecraft.

The simulator will include a detailed analytical model of the spacecraft physical construction, including location of instruments and thrusters. The simulator will be on a computer with interfaces to the instrument in the way the real spacecraft computer will interface. The inputs from the instrument will be used with prototype flight control software to compute the required commands for the thrusters and for the instrument. The effect of the thruster firings will be calculated to infer the effect on the spacecraft attitude and position. These changes will be fed into mechanical actuators that will move the instrument in a manner that simulates the spacecraft motion.

An engineering model of the instrument of one LISA spacecraft will be constructed. This will include full scale versions of the actual sub-systems. Bread-board versions of high-risk sub-systems that have been developed and tested will be included. Simpler version of items that can be procured with existing technology will be included, such as the telescope and mechanical assembly. The engineering model will be placed on a 6-axis platform that can be moved in response to commands from the spacecraft simulator. The engineering model will be incorporated into the three-arm interferometry test bed so that operation of the engineering model can be tested in a simulated spacecraft environment.

3. Technology Development Objectives

3.1 Inertial Sensor

Each LISA spacecraft will carry two inertial sensors. These sensors comprise a free falling proof mass, an enclosing housing with capacitor plates for sensing and forcing, a discharging subsystem, a caging mechanism, a venting mechanism and associated electronics.

The proof masses will define the ends of interferometer arms for the purposes of the primary scientific measurement. The design of the inertial sensor will limit unwanted disturbances from stray forces on the proof mass to an acceptable level (i.e., less than that caused by gravitational waves in the frequency range 10^{-4} Hz to 3×10^{-3} Hz). The inertial sensor will provide and displacement information to the spacecraft for drag-free control.

The technology development plan laid out in this section will advance inertial sensor technology from its current state to a satisfactory level of flight readiness for the LISA mission. As a working definition of "a satisfactory level of flight readiness" both the ESA LISA Science Team and the NASA Mission Definition Team are pursuing an inertial sensor and drag-free system flight demonstration, that would demonstrate an acceleration noise performance about an order of magnitude below that needed for a LISA mission, probably within a slightly more restrictive frequency range (see section 3.1.1). The flight demonstration would not be expected to reach the full LISA performance goal, because it is expected to be too expensive for the flight test to meet the spacecraft environmental requirements of the LISA mission. Instead the flight test will include diagnostic experiments to quantify the inertial sensor performance as a function of known disturbances, so that the performance of the inertial sensor under the flight test conditions can be modeled and extrapolated to the expected performance under the LISA environmental conditions.

The desired flight demonstration constitutes a very significant component of this technology development plan for inertial sensors, and requires major resources. Further, since the conclusions from a flight demonstration would have to been known in advance of the desired 2005 Phase C/D start for LISA, and possible flight opportunities could launch as soon as 2003, the inertial sensor technology development schedule is dominated by the pace demanded by a flight demonstration.

The ESA LISA Science Team has developed a flight demonstration concept called ELITE (ELITE 1998). Although the ELITE mission concept was proposed for a European launch opportunity, its payload description is probably representative of any likely LISA flight demonstration. NASA's New Millennium Program has recently formed an Integrated Mission Definition Team to consider a demonstration of the LISA inertial sensors and the disturbance reduction system for possible inclusion on DS-5. Only very preliminary design definition documents are available.

The LISA inertial sensor requirements are laid out in the next section. In the following section, the state of the technology is described, including the heritage relevant to space flight and a candidate design developed specifically for LISA. That design is the product of studies done by both European and U.S. mission teams.

The technology development plan for inertial sensors described below has four main tasks which progress from the development of the design to the production of hardware for a flight demonstration. The first task, Reference Design Development, will begin with the candidate design and produce a fully detailed engineering design suitable for construction. A laboratory model will be constructed and tested in the Prototype Development task. The inertial sensor head of that prototype unit will follow the flight design as nearly as possible, but the electronics will be only functionally correct (e.g., wire-wrap). The Demonstration Flight Unit Development will produce two flight-qualified units for the demonstration mission mentioned previously. The final main task produces mechanical dummies and inertial sensor simulators for the system test bed.

3.1.1 Inertial Sensor Requirements

As the defining end mirror of an interferometer arm, each proof mass of an inertial sensor must have an acceleration noise less than 3×10^{-15} m/s^2/ Hz in the frequency range 10^{-4} Hz to 3×10^{-3} Hz, so that the proof mass displacement due to non-gravitational forces is smaller than that due to the observed gravitational waves. To keep the forces from the spacecraft on the proof mass at an acceptable level, the spacecraft position must be controlled with respect to the proof mass with accuracy of 10^{-9} m/ Hz. The inertial sensor provides the measurements of the position of the spacecraft with respect to the proof mass used for the spacecraft position control. The spacecraft orientation is controlled by pointing measurements provided by the interferometer system. The orientation of the proof mass must be controlled with respect to the spacecraft with a resolution of 5×10^{-8} radian/ Hz so that interferometer signals reflected off the proof mass are properly directed.

An acceleration noise budget has been developed by considering the sources of noise for a specific design, at some level of definition, estimating their magnitude and then allocating a noise amount and a number of such sources to account for both known and unknown noise sources. Treated as acceleration noise sources are those with a frequency dependence like spurious accelerations which dominate from 10^{-4} Hz to 3×10^{-3} Hz.

Table 3.1 gives the noise allocation for sources of acceleration noise, per inertial sensor. Thermal distortions of the spacecraft and payload can give rise to local gravitational forces. Dielectric losses are the dissipation mechanism by which the electrostatic restoring forces on the proof mass introduce fluctuations. Residual motions of the spacecraft introduce gravity noise. Thermal gradients across the proof mass cavity give rise to an imbalance of thermal radiation on the proof mass. The residual charge on the proof-mass gives rise to electrical forces and a Lorentz force from the interaction with the interplanetary magnetic field. The residual gas impacts on the proof mass from variations in outgassing can generate balanced forces. Thermal fluctuations, expected to have a nearly f^{-2} spectrum, in the telescope will result in optical path changes that look like acceleration noise. And finally, there will be a residual coupling of the fluctuating interplanetary field with the proof mass. These noise sources are discussed in much greater detail in the LISA Pre-Phase A Report and in the LISA Payload Definition Document.

To get the effect of the acceleration noise on the gravitational wave sensitivity, the noise from the proof masses at each end of two arms is included. Assuming the noise is uncorrelated, the effective position noise would be twice that due to the acceleration noise indicated in Table 3.1. To compare to the interferometer optical path length noise, another factor of two is needed to account for the round-trip optical path.

Table 3.1 Acceleration noise budget in units of 10^{-15} m/s^2/ Hz at 10^{-4} Hz.

Error Source	Error
Thermal distortion of spacecraft	1.0
Thermal distortion of payload	0.5
Thermal noise due to dielectric losses	1.0
Gravity noise due to spacecraft displacement	0.5
Temperature difference variations across cavity	1.0
Electrical force on charged proof mass	1.0
Lorentz force on charged proof mass	1.0
Residual gas impacts on proof mass	1.0
Telescope thermal expansion	0.5
Magnetic force on proof mass from fluctuating interplanetary field	0.5
Other substantial effects	0.5
Total Effect of Accelerations (for one inertial sensor)	3.0

3.1.2 State of the Technology

Inertial sensors based on capacitive sensing of free-falling proof masses have been flown in drag-free disturbance reduction systems for more than two decades. The design is also very closely related to space accelerometers which have a considerable technical heritage. The accelerometer mode of operation requires achieving the highest possible precision in measurement of the proof mass, leading to an optimal balance between the readout noise and acceleration noise introduced by the readout system. For an inertial sensor, the precision requirement of the position readout is relaxed, allowing a lower level of acceleration noise. For an accelerometer there are additional requirements on calibration, linearity and sensitivity of the capacitance readout not relevant to inertial sensors. The present inertial sensor technology benefits from a considerable technical heritage, but the acceleration noise required for LISA is still substantially below what has been demonstrated.

3.1.2.1 Technical Heritage

Lange (1964) made the first comprehensive analysis of a drag-free satellite. In 1972, the TRIAD mission demonstrated that a disturbance reduction system based on capacitive sensing of a spherical proof mass and intermittent thruster firing could reduce the long term acceleration noise from strongly varying atmospheric drag and solar radiation pressure in an 800 km earth orbit to 5×10^{-11} m/s^2 or less. That disturbance reduction system was built jointly by Applied Physics Laboratory and Stanford University. TRIAD was followed by the Nova series of navigation satellites that utilized a single degree-of-freedom disturbance reduction system.

In the 1970's, the Office National d'Etudes et de Recherches Aerospatiales (ONERA) began the development of a series of accelerometers based on capacitive sensing of proof masses which continues to this day. The CACTUS accelerometer flew aboard the CASTOR-D5B satellite from 1975 to 1979. The GRADIO accelerometer, based on a 4 x 4 x 1 cm proof mass, was developed for ESA's ARISTOTELES mission but never flown. Differential ground tests demonstrated noise levels of roughly 10^{-8} m/s^2/ Hz down to about 25 mHz. ONERA flew the ASTRE accelerometer, a design very similar to GRADIO, on STS78, STS83 and STS94, but the noise environment on the Shuttle Columbia limited the sensitivity of the test below that in the ground demonstrations.

ONERA has two scheduled flights for their evolving line of accelerometers. The STAR accelerometer is scheduled to launch on the German geodetic satellite CHAMP in 1999. The SuperSTAR accelerometer is scheduled to launch on NASA's gravity field monitoring mission GRACE in 2001. In the latter mission, the accelerometer resolution is targeted to be 10^{-10} m/s^2/ Hz at 5 mHz and above.

Other inertial sensor and drag-free technologies have been studied and developed for tests of inertial frame dragging and the Equivalence Principle. Gravity Probe-B (GP-B), now in the final stage of assembly at Stanford, uses drag-free technology to minimize the force applied to the gyro rotors. One of the spherical rotors is used as the proof mass and its position is sensed electrostatically. The acceleration noise on a rotor <u>not</u> used as the inertial sensor is expected to be 2.6×10^{-10} m/s^2/ Hz from 2-20 mHz, due to noise in the control voltages. This performance is actually a compromise to minimize torques on the proof mass. One might expect the acceleration noise in the inertial sensor rotor to be considerably lower. The various concepts for a Satellite Test of the Equivalence Principle have all included the use of one of the cylindrical proof masses as an inertial sensor for drag-free operation of the satellite. Much of the technology is derived from GP-B, but there isn't an obvious relevant performance expectation.

In summary, the LISA acceleration noise requirement is several orders of magnitude lower than has been demonstrated, and at somewhat lower frequencies. The LISA requirement differs from more conventional drag-free applications in that only noise on the proof mass matters. Residual noise on the spacecraft does not. It should also be noted that the LISA mission concept places the inertial sensor in a far more benign environment than encountered by any ground test or earth orbiting missions. For example, the LISA spacecraft will experience constant solar illumination; the direction to the Sun moves about a cone with a 30° half angle and a one year period; and the inertial sensor should experience thermal disturbances at or below 1 μ°K/ Hz in the measurement band.

The most significant component of the technology development of inertial sensors will be the demonstration of a disturbance reduction system in a test flight with a suitable environment.

3.1.2.2 Candidate Design

The most highly developed design of the inertial sensors for LISA has been done by ONERA (Pre-Phase A Report, Payload Definition Document, Rodrigues and Touboul 1998), based on their accelerometer technology, and is shown in Fig. 3.1. A sensor consists primarily of a $35.4 \times 35.4 \times 51.0$ mm proof mass of gold-platinum alloy, a ULE housing with gold coated electrodes, a ULE or invar sole plate for attachment to the optical bench, a caging mechanism, a titanium vacuum enclosure and a charge control system based on a UV lamp (not shown). [Note that the charge control system is not part of, but is accommodated by, the ONERA design. However, it is a subsystem of the inertial sensor in the LISA baseline design and is treated as such in this document.] The design utilizes overhanging electrodes with a gap of 1.5 mm in the direction of the interferometric measurement, and conventional electrodes with gaps of 0.3 mm for the other degrees of freedom. The overhanging electrodes sense capacitance changes caused by changes in overlap area rather than changes in gap.

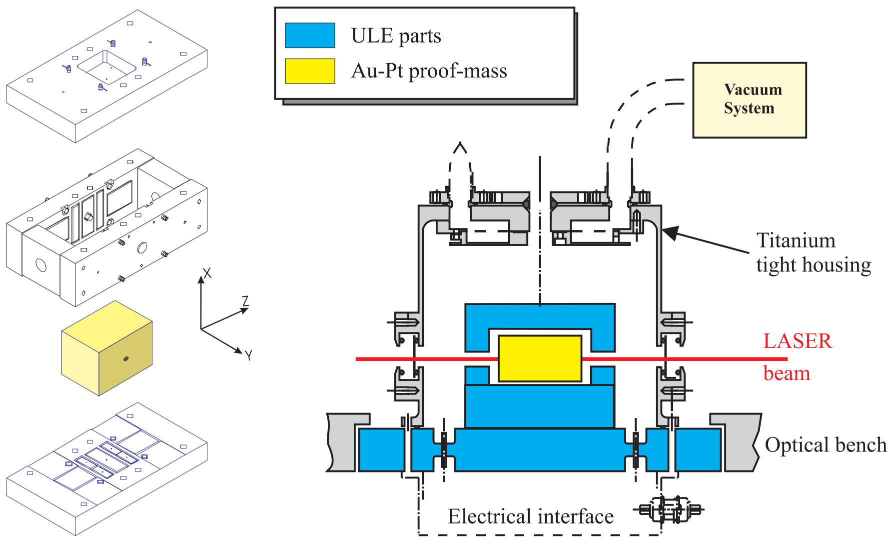

Figure 3.1 ONERA's CAESAR, a preliminary design for LISA.

There have been analyses of variations on the ONERA design (e.g. Speake and Andrews 1997; Vitale and Speake 1998). Design discussions within the LISA science teams have explored different proof mass aspect ratios, electrode geometries, electrode gaps, discharging schemes and caging mechanisms.

The baseline design of two inertial sensor subsystems, caging and charge control, need further definition. The GP-B gyros employ a caging mechanism based on a pneumatic piston that traps the rotor against the housing wall. A suitable proof mass caging mechanism has not been employed in the ONERA accelerometer test flights to date, but some design concepts have been developed. A mechanism with actuator for LISA needs to be developed and demonstrated in the lab. A design that clamps the proof mass in the middle of the housing is desirable. The proof mass must be held safe against launch vibration loads, and the mechanism needs to not interact with the proof mass through parasitic forces after being released.

Cosmic rays can charge the proof mass and its housing, at a rate of about 10^{-17} to 10^{-18} C/s, to unacceptable levels. An active charge control system must sense the charge of proof mass and housing and reduce it to acceptable levels ($<2\times10^{-14}$ C). In the baseline LISA design, sensing is achieved through the capacitor plates, and the charge is reduced by UV photoemission with light delivered by a fiber from a low voltage mercury lamp. Although this scheme has been used in GP-B (Buchman et al. 1995), some details of a UV-lamp based charge control system compatible with the LISA inertial sensor design needs to be worked out. A similar charge control system has been developed and flight qualified at Imperial College.

DeBra (1998) and others at Stanford University have begun to contemplate alternate design concepts for the LISA inertial sensor based on the GP-B technology and on developments for the Satellite Test of the Equivalence Principle (STEP). A spherical proof mass could have several advantages, such as requiring only one inertial sensor per spacecraft rather than the two in the baseline design, eliminating over constraint and the need for forcing on proof masses. However, the need to make and adequately smooth sphere and the need to spin the mass introduce additional challenges. An optical rather than capacitive readout would reduce the electrostatic stiffness between the housing and the proof mass but might increase the complexity of the sensor.

3.1.3 Reference Design Development

The first major task for inertial sensor technology development is the formulation of a detailed engineering design. This task has three sub-tasks which (1) examine conceptual level design trades, (2) model the noise and performance of an inertial sensor design and the disturbance reduction system of which it is a part, and (3) complete a detailed engineering design. The task concludes with a review of the design and the readiness for prototype construction.

3.1.3.1 Design Trade Studies

There are two levels of design choices to be made. At the higher level, the applicability of some GP-B and STEP design features have to be evaluated. At present, there are some promising features in those technologies, but no preliminary design applicable to LISA has been worked out. Designs based on those GP-B and STEP features need to be developed, analyzed and compared with the current baseline. The main issue is the shape of the proof mass, spherical or parallelepiped. However, there are other issues such as fabrication techniques to be considered as well. One possible outcome is a merging of the ONERA and Stanford technologies. Additionally, the use of an optical, rather than capacitive, readout needs to be evaluated, again requiring a preliminary design concept. These design trades have far reaching consequences, and will influence other mission systems such as spacecraft attitude and position control. The schedule demands of the DS-5 flight demonstration may require the higher level design choices be made in 1999.

The lower level of design choices that have to be traded involve optimization of parameters and materials. These can only be made in the context of a particular higher level choice. The specific trades to be addressed are: electrode configuration, the electrode gaps, the housing material and the electronics design. The choices of electrode configuration are overhanging versus parallel plate. The CAESAR design (Figure 3.1) utilizes an overhanging electrode (where the overlap is sensed) along the sensitive direction and parallel electrodes (where the gap is sensed) for the other degrees of freedom. The gaps between the electrodes and the proof mass are a compromise between signal-to-noise ratio in the capacitance detection and acceleration noise from variation in work function. The ONERA design makes preliminary choices for these parameters. Advantages of a metal housing have to be compared to those for a ULE housing. The optimum design of the capacitance measuring electronics should be determined. Work at the University of Birmingham has suggested that there may be signal-to-noise advantages in a different capacitance measuring scheme. The caging and charge control subsystem designs will be worked out at a conceptual level under this task. The lower level design trades do not have much impact beyond the inertial sensor, and can generally be later in the development cycle. This sub-task includes developing the analysis to make these design trade choices, and consequently is coupled to following one. Models will be used to assess design choices, perform design trades and optimize design parameters.

The product of this sub-task is a conceptual design, including specification of critical materials operational parameters. The geometry of the proof mass, the readout scheme, electrode configuration, capacitor gaps, proof mass and reference housing materials, and caging scheme will all be specified.

3.1.3.2 Modeling

Substantial noise and/or performance calculations have been done at ONERA and the Universities of Birmingham, Colorado and Trento. The noise calculations typically compute the effect of each noise process separately (see Table 3.1 above, Rodrigues and Touboul 1998, Vitale and Speake 1998). The performance calculations combine the results for many noise sources, sometimes allowing for unmodeled errors. This sub-task involves: collecting noise calculations, comparing their results for the same input, making them uniform with respect to their assumptions and calculations, combining the noise calculations into a performance calculation, coupling these to a model of the dynamic behavior of the charge control system and the disturbance reduction system (i.e., the drag-free control system), and optimizing design parameters. There need to be versions of the models for prototype units, demonstration flight units and the science mission units, since there will be subtle differences in the noise models for each application. The flight units will also have lightweight masses for levitation in a 1 g environment. The models for the flight demonstration and the full LISA mission will differ in the noise effects brought in by different optical benches and different disturbance reduction systems (e.g., different spacecraft).

The product of this task is an end-to-end model of an inertial sensor where the effects of external disturbances (e.g., thermal and electromagnetic), internal noise and the response of the inertial sensor and disturbance reduction system on the acceleration noise of the proof mass are calculated. Of necessity, only a simplified dynamic model of the drag-free control system can be constructed prior to the detailed definition of spacecraft and propulsion parameters. The models developed under this task will likely have to be incorporated into the drag-free control system model and a software spacecraft simulator for both flight demonstration and LISA science missions.

This sub-task is coupled to the preceding one. Noise models and the performance calculations are dependent on both levels of design choices. Specific input to models, for example, thermal disturbances, are dependent on spacecraft and payload design choices. This will require some level of flexibility in the models and ongoing interaction with demonstration and science mission planning.

3.1.3.3 Detailed Engineering Design

The conceptual design produced by the work of the two previous sub-tasks must be rendered into an engineering design suitable for construction. This sub-task will complete the definition of the inertial sensor design so that it can be fabricated. The principal activities will be the mechanical design of the sensor head (i.e., proof mass, reference housing, caging subsystem, charge delivery components, and vacuum system), the design of the electronics (i.e., position sensing circuits, charge sensing circuits, electrostatic forcing circuits, caging control and actuation, UV intensity control circuits and lamp power supplies), and the specification of the payload interface. The baseline design of the charge control and caging subsystems for the LISA inertial sensor will be defined in this sub-task. The final design will specify tolerances and fabrication methods where appropriate. The product of this sub-task will be a detailed description of the reference design.

3.1.3.4 Conceptual Design Review

At the conclusion of the sub-tasks of the Reference Design Development, a conceptual design review is planned. This review will examine whether all reasonable design trades have been productively explored, whether the reference design is compatible with LISA's scientific objectives, whether the construction of the reference design is feasible for both the demonstration and the science flights and whether the program is ready to construct a prototype. The product of this review will be a report.

3.1.4 Prototype Development

The evolution of inertial sensors leads through two development models to the final flight model. The first development model will be a prototype with a sensor head made to the flight design and functional electronics. The goals are to validate that the components of the sensor head can be constructed to design specifications, to demonstrate their functionality and to demonstrate that the electronics design performs as anticipated. The prototype will be used to develop the caging and the charge control subsystems and demonstrate their operation in a prototype LISA head. Functional testing of the prototype will include uncaging, recaging, charge sensing, discharging and position sensing in some degrees of freedom. The mass will not be levitated. It is anticipated that the caging system will have precise position control of the proof mass so that it can be moved about to exercise the capacitive position sensing. Control software for sensor operations will be developed and tested.

3.1.4.1 Proof Mass Fabrication

The proof mass material in the candidate design is a gold/platinum (73%/27%) alloy which can, theoretically at least, have a vanishing magnetic susceptibility, but practically may be difficult to make. Precision cylinders of alloys approaching this mix are made commercially for use as mass standards around high magnetic fields. For LISA, the important technical questions to be answered by this sub-task are: how low a susceptibility can be practically achieved, can the material be machined into the desired shape with the desired tolerances, what are appropriate caging techniques to preserve the gold coating. The product of this sub-task is a proof mass for the prototype unit satisfying the reference design specifications, as well as fabrication processes and handling techniques suitable for producing other test masses.

3.1.4.2 Reference Housing

In the candidate design, the reference housing is constructed of ULE plates sitting on an invar sole plate with gold-deposited electrodes and grounding surface. UV fibers and the caging mechanism are attached to it. A charge generation surface and possibly limit stops for the proof mass will be machined into the housing. The reference housing needs to be a stable, precision structure. The goal of this sub-task is to develop and demonstrate the process for fabricating the housing components, coating the electrodes and grounding surface, and assembling them. The product of this sub-task is a reference housing for the prototype unit satisfying the reference design specifications, and documented fabrication and assembly processes for producing other reference housings.

3.1.4.3 Caging Mechanism Development

A caging mechanism consistent with LISA requirements will be developed. A design concept will be selected as part of the Detailed Engineering Design (section 3.1.3.3), possibly from either the ONERA or GP-B heritage, and then implemented on the prototype unit in this sub-task. It is highly desirable in an inertial sensor to be able to demonstrate the operation of the position sensing subsystem. The TRIAD drag-free satellite used the caging mechanism to move that spherical proof mass along a path that exercised all three degrees-of-freedom in a controlled fashion to demonstrate functionality of the readout system through telemetry at the launch site. Having the capability to position the proof mass at the sensitivity level of the capacitive readout, and withdraw the mechanism may be too complex and difficult to do. The goal of this sub-task is to build the caging mechanism, actuator and controller called out in the reference design. Control software for the caging subsystem to run in the payload computer will be designed, written and tested in this sub-task. The product is a caging subsystem for the prototype unit.

3.1.4.4 Charge Control Subsystem Development

The components of the charge control subsystem described in the reference design will be fabricated. These components include two UV lamps, their power supply, a means for intensity modulation, delivery fibers, and an electronic interface for error signal input. The proof mass charge is measured by applying a rapidly varying voltage to a special set of capacitor plates on the reference housing. The charge of the proof mass relative to the reference housing is read out by synchronous detection of the motion of the proof mass transverse to the measurement direction. This charge sensing function is realized by the components described in other sections. The charge measurement is read out of the sensor by the payload computer, or its surrogate in the case of the prototype unit, and processed to form an error which is sent back to the charge control subsystem interface. Control software for the charge control subsystem to run in the payload computer will be designed, written and tested in this sub-task. The output ends of the fibers have to be mounted on the reference housing, and hence must feed through the vacuum housing. The goal of this sub-task is to build the charge control subsystem components according to the reference design. For the purposes of the prototype, only the delivery end of the fibers need to conform to flight design; the lamps, power supplies, intensity control and electronic interface could be breadboards. However, depending on the similarity of the design to existing space-qualified, UV lamp-based discharge systems such as that for GP-B, it may be desirable to construct other components with flight qualifiable materials and methods to demonstrate feasibility. The product is a charge control subsystem for the prototype unit.

3.1.4.5 Vacuum Housing

The LISA inertial sensors are expected to be assembled in a clean environment, enclosed in a vacuum shell, evacuated or back filled with an inert gas (e.g., dry nitrogen), and maintained in the vacuum housing until sometime after launch. In the candidate design, the vacuum shell, together with the sole plate, define the mechanical envelope of the sensor head, and the mechanical, electrical and optical interfaces with the payload. The goal of this sub-task is to construct the vacuum housing with the associated tip-off port for pump down and/or back filling, the in-flight venting valve, optical windows for the laser beam to reach the proof mass, the seal to the sole plate, and all electrical and fiber connections/feed-throughs. For the purposes of the prototype testing, the vacuum housing permits vacuum operation of the components within. The in-flight venting valve, a frangible disk in the candidate design, will have to be supplemented with a re-closable valve for lab testing. A pumping station and back-filling apparatus will be required as part of the prototype unit support equipment. The product of this sub-task is the vacuum housing for the prototype unit.

3.1.4.6 Electronics

Since the electronics for the LISA inertial sensors is expected to be readily reproduced in a flight qualifiable form, only breadboard construction is anticipated for the prototype unit. The goal of this sub-task is to construct functional versions of the position sensing electronics, the charge sensing electronics, and the electrostatic forcing electronics. The position sensing electronics consists of preamplifiers and bridge circuits for each plate. Regardless of the capacitance measuring circuit, it is expected that a voltage representing the measured capacitance will be digitized and read by the payload computer. The charge sensing electronics includes an oscillator, a drive amplifier and the synchronous detection circuits with digital interface. The electrostatic forcing electronics, which apply forces to control the orientation of the proof mass and possibly a DC component to counter a bias from the spacecraft environment, consist of a digital interface to the payload computer, D/A conversion and a drive amplifier comprise. Construction of the electronics for caging control and actuation, UV intensity control circuits and lamp power supplies has been covered in previous sub-tasks. All cables connecting to the sensor head at the vacuum housing are expected to use flight connectors. The product of this sub-task is breadboard electronics for position and charge sensing and electrostatic forcing.

3.1.4.7 Assembly

In this sub-task the products of the previous six sub-tasks are assembled into the complete prototype inertial sensor. The reference housing will be assembled around the proof mass and the caging mechanism will be installed in the reference housing, which in turn will be attached to the sole plate. Electrical and fiber connections from housing to sole plate will be made. The vacuum housing will be closed, and the volume will be pumped and back filled. Electrical and fiber connections from the vacuum shell/sole plate will be made to the external electronics units. The product of this sub-task is a fully assembled prototype inertial sensor ready for functional testing.

3.1.4.8 Functional Testing

In this sub-task the operational functions of the prototype will be tested. Those functions are: vacuum venting, uncaging, re-caging, positioning of the proof mass with the caging system, electrostatic sensing of the proof mass position, charge sensing, discharging and electrostatic forcing. The last three functions will not be tested at the performance levels required by LISA in a 1 g environment, nor will they be tested under closed loop control. The position sensing will be tested by moving the mass with the caging mechanism and reading out the capacitance changes. The charge sensing will be tested by injecting varying amounts of charge on the proof mass and moving it synchronously with the caging mechanism, a model dependent method. The discharging will be tested by varying the relative charge on the reference housing and the proof mass using the UV lamps and sensing the relative charge through the caging mechanism. Depending on the caging actuator, it may be possible to measure the electrostatic forcing through the compensating force required of the caging mechanism. The functional testing will require a surrogate payload computer and test programs to be developed under this sub-task. The product of this task will be validation of the functionality of the prototype unit.

3.1.4.9 Acceptance and Critical Design Review

At the conclusion of the Prototype Development task, a review will be held to evaluate the construction lessons, the test results and the suitability of the reference design, as amended by the prototype activities, for constructing demonstration flight units. This review will judge whether the functional tests were adequate and the results were sufficiently convincing to accept them as a demonstration of the suitability of the design, to the extent possible on the ground. It will also judge whether the reference design, augmented by the construction, assembly and testing knowledge gained with the prototype experience warrants advancing to the next generation of development units, the models for the demonstration flight. The product of this review will be a report with (1) recommendations for changes in the reference design and (2) a clear statement of approval to proceed or recommendations for remedial action.

3.1.5 Flight Demonstration Unit Development

This major task will produce two flight units for a technology demonstration mission. Demonstration of inertial sensor performance will be a critical component of a LISA flight demonstration. The acceleration performance will be validated through interferometric comparison of two inertial sensors.

The inertial sensors will be designed and constructed to reach the LISA performance goal. On the flight test, the LISA performance goal is not expected to be reached, due to looser tolerances on the flight demonstration spacecraft. The flight-demonstration inertial sensors will include additional coils, heaters, etc. to introduce known perturbations into the environment so that the inertial sensors performance model can be validated. The flight demonstration will validate the inertial sensor performance to with an order of magnitude of the desired performance for LISA, and the modeling experiments will show how a better-designed and controlled spacecraft environment will allow the LISA performance goal to be attained.

The Flight Demonstration Unit Development consists of several sub-tasks: the fabrication of the subsystems, their assembly, ground tests, integration support and actual flight. The schedule is based on launch in late 2003, a 6 month flight and analysis to be complete by the end of 2004. Although outside of the scope of this task, a critical design review and design modifications, if any, could be completed for a LISA Phase C/D start in 2005.

3.1.5.1 Proof Mass

This sub-task will produce two proof masses for the demonstration flight units. The masses will be specified by the reference design, and manufactured by the process developed in the prototype construction.

3.1.5.2 Lightweight Proof Mass

It is desirable to check the performance of the LISA inertial sensors on the ground. However, it will be impossible to levitate the proof masses in the Earth's gravity using the electrostatic suspension technique used in accelerometers, due to the large mass and lower electrical coupling for the LISA sensor. It may be possible to create a lighter test mass which could be levitated with the LISA sensor geometry. If the identical shape is retained, the material density needs to be $\ll 1$ gm/cm^3. This might be achieved by coating an aerogel test mass. Alternatively it might be possible to create a hollow, or partially hollow proof mass. An effort will be made to develop a suitable light-weight proof mass for testing. If successful, this would simplify requirements on the clamping mechanism for some tests of the functionality of the sensor and electronics.

3.1.5.3 Reference Housing

This sub-task will produce two reference housings for the demonstration flight units. The housings will be specified by the reference design, and manufactured by the processes developed in the prototype construction.

3.1.5.4 Caging Subsystem

This sub-task will produce two caging subsystems for the demonstration flight units. The mechanisms will be specified by the reference design, and manufactured by the processes developed in the prototype construction. The actuator, control electronics and payload interface will be flight qualifiable. The control software will be adapted from the software for the prototype unit.

3.1.5.5 Charge Control Subsystem

Two charge control subsystems will be produced for the demonstration flight units. The mechanisms will be specified by the reference design, and manufactured by the processes developed in the prototype construction. The UV lamps, the power supplies, the intensity control, the entire delivery fibers and the payload interface will be flight qualifiable. The control software will be adapted from the software for the prototype unit.

3.1.5.6 Vacuum Housing

Two vacuum housings will be produced for the demonstration flight units. They will be specified by the reference design, and manufactured by the processes developed in the prototype construction. These units will not have the supplemental re-closable valve included in the prototype units.

3.1.5.7 Electronics

Two sets of position sensing electronics, charge sensing electronics, and electrostatic forcing electronics which can be flight qualified will be produced. These units will have to be configured a mounting in equipment compartments appropriate to the demonstration mission payload. That payload design will be defined outside of this technology development plan.

3.1.5.8 Model verification components

A number of devices will be designed and integrated into the sensor housing to provide calibrated disturbances to the sensor environment. This will include coils for generation of magnetic fields, heaters to create temperature fluctuations, etc. Operation of these disturbance during the flight test will allow verification of the inertial sensor noise models, and allow investigations into the limiting effects during the flight demonstration.

3.1.5.9 Assembly

Under this sub-task, the two flight units will be assembled.

3.1.5.10 Ground Testing

While ground tests cannot reach the low disturbance levels necessary to evaluate a LISA inertial sensor, they can provide a valuable and relatively inexpensive means for exploring technical design issues and for functional testing of flight units. Three kinds of ground tests are possible: functional testing of the sort described in section 3.1.4.8, environmental testing for flight qualification, drop tower testing, and electrostatic levitation. The functional testing will be performed before and after environmental testing. Environmental testing will involve thermal stress, vibration, vacuum and possibly radiation. Both of these kinds of ground tests will definitely be performed. Drop tower testing, for example at the Bremen drop tower, can give zero g environments for up to 5 seconds, and may permit more realistic functional testing of charge sensing. Needless to say, drop tower tests impose costs in design and support equipment, and some risk to flight units. The value of drop tower tests will have to be assessed when the tradeoffs are better understood. Likewise electrostatic levitation will have to be re-assessed when the benefits and costs are clearer. At present neither drop tower tests nor levitation tests seem likely.

3.1.5.11 Flight Readiness Review

At the conclusion of ground testing, a review will be held to determine if the demonstration flight units are ready for flight. This review will judge whether the functional and environmental tests have been adequate to certify the units for flight. The product of this review will be a report with a clear statement of approval to proceed to flight or recommendations for remedial action.

3.1.5.12 Payload, Spacecraft and Launch Vehicle Integration

Personnel knowledgeable about the inertial sensor flight units will support those units during their integration into the demonstration flight payload, and the subsequent integration of the payload onto the spacecraft and the spacecraft onto the launch vehicle. In particular, the support team will work closely with the personnel responsible for the disturbance reduction system. Some of the functional tests will likely be possible at each step. The support team will travel to the site of the integration and finally the launch site to insure the safe transport of the flight units, validate the condition of the flight units and assist the integration teams as necessary.

3.1.5.13 Flight Support

Finally a support team will participate in the demonstration mission, supporting operations and analyzing data on the inertial sensor performance. The flight support team will support the mission operation as required, either remotely or at the operations center. That support will include advice for planning tests, consultation during tests as requested and prompt analysis of test results that pertain to the inertial sensor. The latter activity will involve close interaction with the group responsible for the disturbance reduction system. The performance model developed in the Reference Design Development task will be made current with the final mission design to correctly take into account interactions between the inertial sensors and the payload and the spacecraft. The dynamic model of the disturbance reduction system will be updated to incorporate information about the spacecraft mass and moments and thruster properties. The product of this sub-task will be a final report assessing the performance of the inertial sensor units on the technology demonstration flight.

3.1.6 Inertial Sensor Simulators for System Test Bed

In this final major task of the inertial sensor technology development, simulators will be designed, built, installed and operated in the spacecraft simulator and the system test bed. This will enable investigations in the operation of a full-scale LISA payload. Interactions between the various control loops will also be studied.

The initial units will be simple mechanical ‘dummies’. These will mount into the Optical Bench in the same manner as the real inertial sensors, and provide fixed reflecting surfaces corresponding to the surfaces of the proof masses. These will allow for operation of interferometry portion of the System Test Bed. Much more detailed simulators will be developed that will have movable reflecting surfaces. Ideally these will be very similar to the actual sensors, but with the proof masses mounted on piezoelectric crystals or other actuators. The proof masses then can move in response to commands from the payload computer, and the proof mass position can be read out by the actual sensor electronics, which will then interface with the payload computer. It may be possible to adapt the Prototype Units for this purpose. The degree of simulation will be investigated and developed after the assembly of the Flight Demonstration Units has begun.

3.1.6.1 Mechanical Dummies

This sub-task will produce two mechanical dummies for the optical benches of the Integrated System Test Bed. The dummies will consist of vacuum housings and sole plates made to the reference design and additional optics mounted on a rotation stages as necessary. The vacuum housing and sole plate from the prototype could be used for one set.

3.1.6.2 Simulator System Design

The goal of the simulators design is to produce controllable proof masses, whose position measurements are as similar to the actual capacitance readouts as possible. The expected response of the controlled proof mass to other inputs from the full-scale payload and spacecraft simulator will be computed based on the inertial sensor model. The proof mass actuators will be designed, and the corresponding changes to the proof mass. Changes to the housing needed to accommodate the actuators will be defined. Requirements for the electronics needed to control the actuators, and to respond to signals generated by the inertial sensor electronics and the spacecraft simulator will be defined.

3.1.6.3 Preliminary Design Review

Before proceeding to construction of the simulation units, a preliminary design review will be held. This review will determine whether the reference design is compatible with the requirements of the Spacecraft Control System Simulator and the Integrated System Test Bed, whether construction is feasible and reasonable and whether the program is ready to construct a prototype. The product of this review will be a report with a clear statement of approval to proceed with construction or with recommendations for remedial action..

3.1.6.4 Construction of Simulated Sensors

This sub-task will produce two simulated sensors for the Integrated System Test Bed.

3.1.6.5 Construction of Electronics

The electronics to support two sets of simulated sensors for the System Test Bed will be constructed. These will include drive electronics for the proof-mass actuators, an interface to the inertial sensor electronics, a computer to control the proof mass position and orientation, and two complete sets of inertial sensor electronics. The readout electronics from the prototype could be used for one set. The software for the simulator computer will be developed. The simulator control computer and its software will be configured to interface to the control processor of the Spacecraft Control System Simulator and the Integrated System Test Bed.

3.1.6.6 Test Readiness Review

At the conclusion of the two previous construction tasks, a review will be held to judge whether the simulation units are ready for testing. This review will judge whether the units as fabricated meet the requirements of the test beds, and are ready to be integrated into them. The product of this review will be a report with a clear statement of approval to proceed with integration or with recommendations for remedial action.

3.1.6.7 Test Bed Integration and Support

The signal simulators will be integrated into the Integrated System Test Bed. A simulator support team will support the operations of the Integrated System Test Bed as required. That support will include advice for planning tests, consultation during tests as requested and prompt analysis of test results that pertain to the inertial sensor. The dynamic model of the disturbance reduction system will be incorporated into the disturbance reduction simulation in the controller of the Spacecraft Control System Simulator. The product of this task will be a report comparing the performance of the simulators with expectations.

3.1.8 Inertial Sensor Schedule and Budget

The schedule for the inertial sensor development is given in Figure 3.2. The milestones are shown in Figure 3.3. The budget required to perform the development is given in Table 3.2.

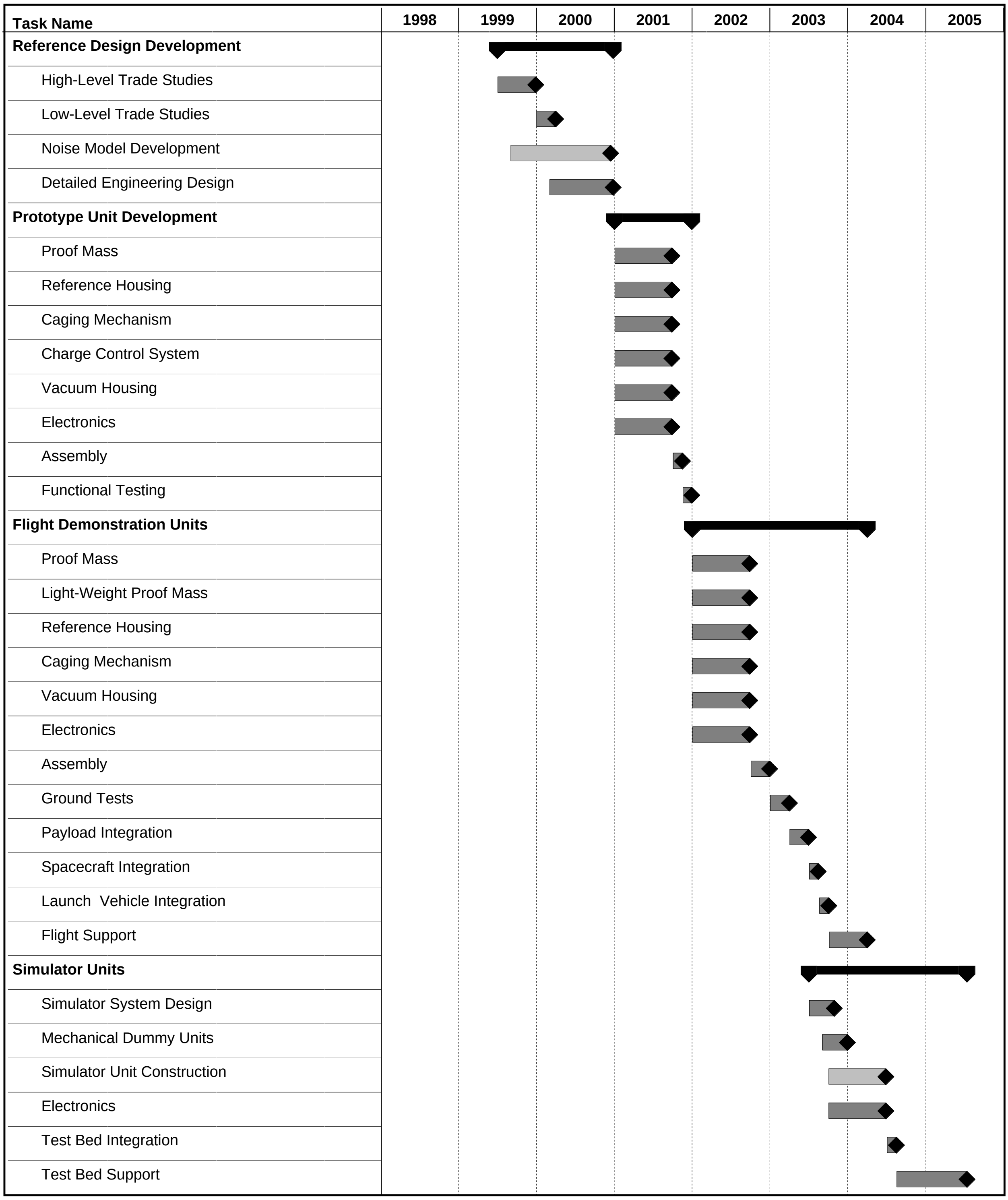

Figure 3.2 Inertial sensor development schedule

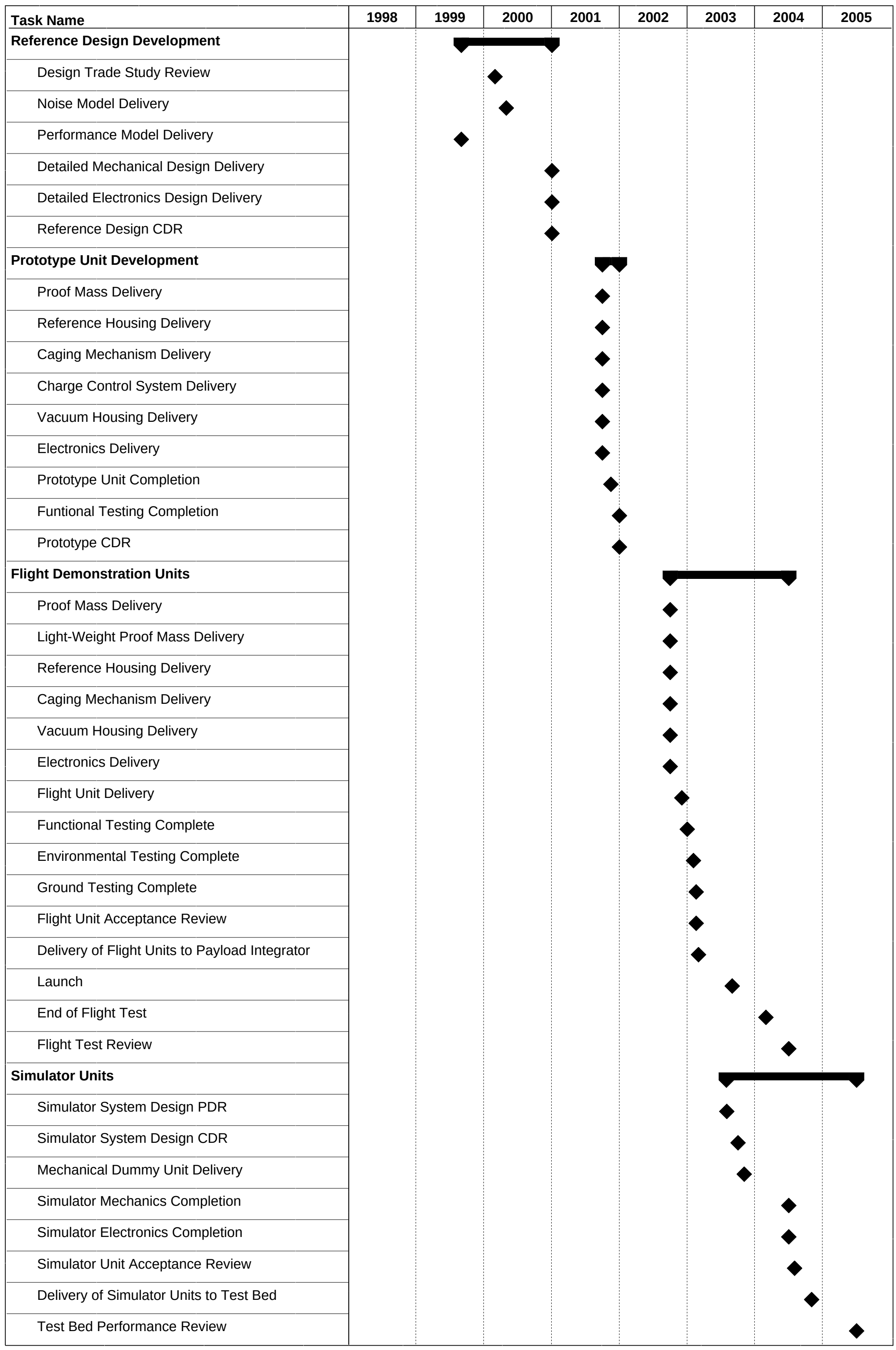

Figure 3.3 Inertial sensor development milestones

Table 3.2 Inertial sensor development budget

Task / FY funds ($k)	1999	2000	2001	2002	2003	2004	2005	Total
Reference Design	**50**	**450**						**500**
Design trade studies	50	150						200
Modeling		100						100
Detailed engineering design		200						200
Prototype Development			**2300**					**2300**
Proof mass fabrication			300					300
Reference housing			400					400
Caging mechanism development			300					300
Charge control system			200					200
Vacuum housing			200					200
Electronics			400					400
Assembly			200					200
Functional testing			300					300
Demonstration Flight Units				**3800**	**1400**	**300**		**5500**
Proof Mass				400				400
Light-weight proof mass				300				300
Reference housing				500				500
Caging subsystem				300				300
Charge control subsystem				500	500			1000
Vacuum housing				300	.			300
Electronics				600				600
Assembly				400				400
Ground testing				500	300			800
Integration support					300			300
Flight support					300	300		600
Simulators for Test Bed					**1100**	**800**	**300**	**2200**
Simulator system design					300			300
Mechanical dummies					400			400
Construction of simulated sensors					200	300		500
Construction of electronics					200	200		400
Test Bed integration						150		150
Test Bed support						150	300	450
Total	**50**	**450**	**2300**	**3800**	**2500**	**1100**	**300**	**10500**

3.2 Micronewton Thrusters

Section 3.2.1 below summarizes the propulsion system requirements for LISA as they are currently understood. Some of these are solely projections based on the current concept for the spacecraft and anticipated mass, power, and volume allocations. These are listed as guides in understanding the various technology verification activities being proposed but could of course change as the mission is further defined. Section 3.2.2 describes the principles of operation of the FEEP ion thrusters.

Section 3.2.3 gives an overview of the ongoing activities and future plans for the ESA-funded thruster development effort. Section 3.2.4 describes a detailed series of investigations to validate the thruster performance and lifetime. The selection of tests has been designed to enhance our understanding of basic thruster physics as they relate to performance and lifetime. The tests will assess thruster system performance as well as potential failure modes, and enable investigation of lifetime performance using accelerated testing and modeling to simulate conditions expected during actual operation. The emphasis in the proposed work is in the area of plume characterization, the physics of the neutralization process and erosion processes.

Section 3.2.5 describes a proposed accelerated service life demonstration test. This test occurs at the end of the roughly four year program and is designed to test as close to a flight-like system as possible under conditions that are most representative of the mission requirements as we understand them at that time. The design of this test in terms of operating parameters and diagnostics will reflect all that has been learned of life limiting mechanisms in the preceding performance and beam characterization tests.

3.2.1 Thruster requirements

The micronewton thruster requirements are listed below in Table 3.3. The precision thrust performance requirements are based on spacecraft position control needed to satisfy the inertial sensor performance requirements. The coarse thrust requirements are based on a preliminary estimate of the amount of spacecraft roll induced at separation from the spacecraft's propulsion module, and the amount of time provided by an assumed battery mass to recover from the tip-off roll. The thruster specific impulse requirement is based on a value that would yield a propellant mass no more than the assumed thruster mass; the nominal thrusters have much higher specific impulse. The mass, power, and volume requirements are based on preliminary values used for the overall mission design.

Table 3.3 Micronewton thruster requirements

Requirement	Value	Comment
Precision thrust range	5 - 25 µN	Oppose solar radiation pressure
Precision thrust control	+/- 0.1 µN	Spacecraft control to 10 nm
Coarse thrust range	25 - 100 µN	Spacecraft tip-off recovery
Coarse thrust control	+/- 1 µN	Spacecraft tip-off recovery
Specific impulse	> 500 s	Keep fuel within mass margin
Lifetime @ 25 µN	>3 yr	10 yr goal
Mass	< 10 kg	4 thrusters+electronics
Power	< 5 W	1 thruster @ 25 µN
Volume	< 1000 cm^3	4 thrusters+electronics

3.2.1 Micronewton metal-ion thruster operation principles

The metal-ion thrusters operate by accelerating ions in an electric field, and ejecting them to develop the thrust (Bartoli et al. 1984). The ions are generated by exposing a free-surface of liquid metal (cesium or indium) to an electric field.

The shape of this liquid surface is established by the counteracting forces of surface tension and electric field stress along a knife-edge slit with a width of about 1µm, or at a Tungsten needle with a tip radius of 2 to ~15 µm. With an applied voltage between 5 and 10 kV, the ions are ejected at a velocity in the range of 60 to ~100 km/s, depending on the propellant and the applied voltage. The mass flow is very low, so the developed thrust is in the desired micro-Newton regime.

By smoothly varying the applied voltage, the thrust can be correspondingly controlled, as desired, all the way down to fractions of a micro-Newton. The FEEP thrusters require less than 5W to develop a thrust of 25 µN. The total propellant (cesium or indium) mass required for the nominal two-year mission is only a few grams per thruster.

There are two designs currently being pursued at Centrospazio (Italy) and at the Austrian Research Centre Seibersdorf. The FEEPs originally developed by Centrospazio were designed for thrust levels in the milli-Newton regime, as required for communication satellites. For LISA, they had to be scaled-down by a factor of a thousand from the original design. The Indium Liquid-Metal Ion Sources (In-LMIS) under development at Seibersdorf were originally designed for spacecraft charge control and mass spectrometers. They have already been flown on various missions and proven their reliability in space during more than 800 hours of operations.

Figure 3.2.1a shows a schematic of the FEEP thruster. The metal reservoir contains cesium which is molten and drawn by capillary action to the end of a slit (or needle). There atoms are spontaneously ionized and accelerated by a high voltage. A separate cathode or filament is used to emit electrons to neutralize the beam and prevent spacecraft charging.

Figure 3.2.1b shows a performance curve from an In-LMIS thruster. The thrust, shown in the bottom curve, is derived from the measured voltage and current in response to a sinusoidal control current. This shows that the thruster is operating in the range required by LISA with a smooth range of thrust response.

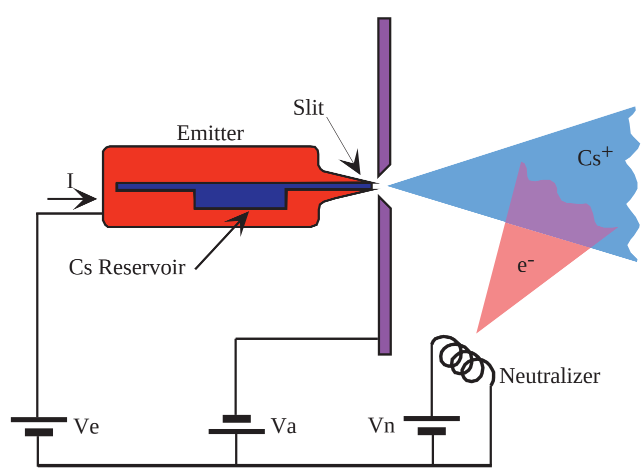

Figure 3.4 Schematic drawing of FEEP thruster.

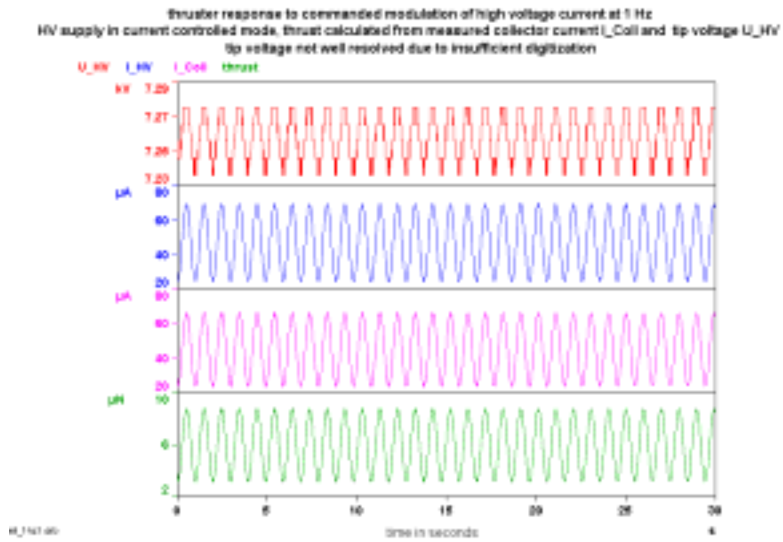

Figure 3.5 Operation characteristics of In-LMIS thruster. The thruster is run in current-controlled mode. The voltage of the supply U_HV (top curve) is self regulating according to the commanded current. The high voltage current I_HV (second curve), produced by the high-voltage supply, controls the emission rate. The beam current I_Coll, which leaves the spacecraft, is shown in the third curve. The thrust (fourth curve) is calculated from U_HV and I_COLL. (courtesy M. Fehringer, Seibersdorf)

3.2.3 Current and planned ESA thruster development

3.2.3.1 Cs FEEP emitter development

ESTEC is funding the development of the Cs FEEP thruster for several potential users which include not only the low thrust scientific mission base but small commercial satellite attitude control as well. Drag free missions are anticipated to require thrust levels on the order of tens of micro Newton whereas larger satellites using FEEPs for more traditional ACS functions are likely to require thrust of several milli-Newton. This thrust range, which spans two orders of magnitude, defines a dual development path for this technology with different requirements and test priorities.

Preparations for a lifetime demonstration test of the FEEP thruster are currently underway. This test will be performed at the ESTEC electric propulsion laboratory starting in September of 1998. During this test the emitter will be operated continuously at a thrust level of 1- 50 µN for 1500 hr. This thrust level corresponds to emission currents in the range of 10 – 500 µA. Diagnostics for this test will include electrostatic probes to characterize the beam. In addition, a Quartz Crystal Microbalance (QCM), witness plates, and a solar cell will be used to characterize deposition of sputtered material. The emitter will use an integral Cs reservoir with a capacity of approximately 4 grams. The neutralizer to be used in the life test is a barium-calcium-strontium oxide, low work function thermionic emitter designed to operate at approximately 600 C with a specific power consumption of 0.6 W/mA.

ESTEC is supporting test and development of the Cs FEEP thruster at Centrospazio in Pisa, Italy. Currently, the highest priority in the FEEP development activity at Centrospazio is preparation for a Shuttle “Get-Away-Special” (GAS) flight demonstration planned for April 2000. The Electric

Microthruster Test in Space (EMITS) experiment will consist of two Cs FEEP thrusters (including neutralizer) one operating at a thrust level of 1 – 30 μN, the other at 1 mN. This test will demonstrate, among other things, microgravity operation of the capillary feed system and operation of the thruster in the water and oxygen environment of the shuttle. Four moveable electrostatic probes and a QCM will provide data on plume current distribution and back-flow contamination. ESTEC is funding Centrospazio for the flight experiment assembly, integration, and launch support. The thruster power and control electronics are being developed by LABEN (Milan, Italy) and the computer control/data storage unit by Techno System Developments (Naples, Italy). This will be the first flight test of a Cs FEEP thruster.

3.2.3.2 In-LMIS emitter development

The indium liquid metal ion source (In-LMIS) has been under development within the Applied Physics Department at the Austrian Research Center Seibersdorf for roughly ten years. This technology, which has been used successfully for spacecraft potential control and as an ion source for a secondary ion mass spectrometer being flown on the Russian MIR station is currently under evaluation for use as a low thrust ion propulsion device. ESTEC is funding a three year activity at Seibersdorf, scheduled for completion in July of 2000, to perform basic physics research on the In-LMIS and to establish its feasibility as a thruster. This contract also includes the development, manufacture, and testing of breadboard and engineering models of a In-LMIS thruster including neutralizer and electronics.

3.2.3.3 Neutralizer development

ESTEC is currently funding Centrospazio to develop a low power, long life neutralizer. In addition, Centrospazio is currently using its own resources to develop cold cathode field emitters. For both the life demonstration and shuttle flight test, Centrospazio is planning on a low work function thermionic cathode.

3.2.3.4 Power control electronics

ESTEC is currently funding LABEN to develop the FEEP Control Unit (FCU) which will be used in the GAS Flight demonstration. This package is anticipated to have a mass of approximately 5 kg and be capable of controlling a cluster of 4 thrusters.

3.2.3.4 Torsion balance thrust measurement

To date, the thrust performance of low current, liquid metal ion sources operating at the micro-Newton level have been calculated from measured emission currents and accelerating potential. This calculation requires certain assumptions be made regarding the ionization efficiency, charge to mass ratio, accelerating potentials. The need for a reliable direct thrust measurement has been recognized for some time and various activities are underway to address this challenge. ESTEC is currently funding a microbalance development activity at the National Physics Laboratory in the UK which should be ready for use at ESTEC in March, 1999. Centrospazio is independently developing a torsional thrust stand and anticipate data will be presented sometime in 1999. The UK thrust stand will be available for use at ESTEC, Centrospazio, and Seibersdorf.

3.2.4 Thruster system performance validation

3.2.4.1 Thrust magnitude, control, repeatability

The thrust magnitude, control resolution, and repeatability over time will be characterized directly on a calibrated thrust balance. Data will include a detailed uncertainty analysis. The current plan is that these measurements will be made by ESTEC or by organizations in Europe supported through ESTEC grants.

3.2.4.2 Noise

Any liquid metal ion source will emit a finite number of macro particles or droplets at some threshold emission current. The current at which this occurs is a function of the liquid surface tension and emitter geometry. Charged droplets are a source of thrust "noise" (as well as mass utilization inefficiency) which must be characterized. In particular, understanding the magnitude and repeatability of this noise is important. Measurements will be made using a mass spectrometer to assess the droplet content and charge-to-mass distribution over a range of operating conditions.

3.2.4.3 Specific Impulse

The mass flow rate of propellant will be measured and used in conjunction with direct thrust measurements to determine the specific impulse. These direct measurements will be compared with calculated specific impulse using emission current and mass utilization efficiency data. The current plan is that these measurements will be made by ESTEC or by organizations in Europe supported through ESTEC grants.

3.2.4.4 Sputter erosion

A number of low energy ions will exist in the plume which are produced through neutral ionizing (of residual gas in the tank) and charge exchange collisions (with atoms in the metal vapor). The accelerator electrode is typically biased several kV negative of the beam potential which creates the risk of physical erosion of surfaces when ions (initially at a low energy) are accelerated back towards the accelerator surfaces.

The energy distribution of ions in the plume will be measured (Retarding Potential Analyzer) and this data used in conjunction with published sputter yield data to assess the extent to which sputter erosion is a concern. If the long term risk cannot be resolved in an accelerated life test then a model based on a standard probabilistic failure analysis methodology may be proposed to assess the risk and help in the identification of a solution.

3.2.4.5 Metal vapor deposition

Both indium and cesium have small but finite vapor pressure at the temperatures likely to be encountered at the emitter surfaces. Over a 3 - 10 year mission life, this raises the possibility of vapor condensation occurring on insulator surfaces creating possible failure due to electrical shorts.

During performance and life tests, visual inspection through the use of in-situ optical diagnostic methods will be performed to assess condensation over different periods of time. In addition, this investigation will be supported by a Particle-in-Cell/Direct Simulation Monte-Carlo (PIC-DSMC) simulation of the neutral back flow.

3.2.4.6 Neutralizer Current-Voltage Characteristics

The neutralizer-plume coupling voltage and current characteristics will be mapped out over a range of parameters. These parameters will include the physical location of the neutralizer as well power level. In addition, different candidate neutralizer technologies (direct thermionic (filament), indirect thermionic (heated low work function substrate), and cold cathode field emission) will be evaluated prior to any down-select for the mission.

3.2.4.7 Neutralizer Pressure Sensitivity

Because of the extremely low emission currents characteristic of the liquid metal ion sources, neutralization can readily occur as a result of secondary electron emission from the test chamber walls as well as ions in the background plasma (produced from ionization of residual gas). It is therefore essential that the effects of facility background pressure on the neutralization process be understood as early on in the program as possible.

Neutralization tests will be performed in a cryo-pumped, Ultra High Vacuum (UHV) test chamber ($P< 10^{-9}$ Torr) where the effects of background pressure on the neutralizer plume interaction can be characterized. This interaction will be quantified through a systematic mapping of the plasma potential and current density distribution under a range of conditions. This data will further be correlated with simulations performed with the PIC-DSMC model to facilitate interpretation.

3.2.4.8 Thruster-spacecraft Interactions

In addition to its use as an interpretive tool, the PIC-DSMC code will play a vital role in understanding the thruster-spacecraft interaction. Its development is essential to eventually meeting requirements which set forth allowable limits of plume induced effects on the spacecraft and instruments. These effects should include but not be limited to back flow of energetic ions capable of sputtering surfaces, neutral back flow which could condense into films on optical surfaces, and attenuation of laser and inter-spacecraft communication signals traveling through the plume. The development of this simulation tool should proceed in parallel with the thruster performance evaluation, with experimental data used to validate the model, and the model used to guide additional experiments.

3.2.5 Thruster lifetime evaluation

3.2.5.1 Flight-Like System Definition

After a predetermined series of thruster and neutralizer performance tests, a down-select will determine the leading thruster-neutralizer-power processor technology candidate for life testing.

3.2.5.2 Accelerated life test design

Data gathered during performance characterization tests as well as any probabilistic failure analysis will be used to establish test conditions for a representative, accelerated service life demonstration test. This test design should include all available information on possible failure modes understood to that point. The test design will further include a carefully selected diagnostic suite in order to verify understanding of the long term behavior of the complete thruster system.

3.2.5.3 Long term performance evaluation

Two key goals of the life test will be 1) characterization of thrust performance (magnitude, controllability, and repeatability) over time and 2) identification of any failure modes that may only become apparent in a long term life test so that preventive measures can be taken. Ideally, the first goal would be accomplished through direct measurement with a thrust balance. If this proves to be unfeasible then thrust performance will be calculated from emission current and charge-to-mass distribution data. In addition to current-voltage data collection, the life test diagnostic suite should include the capability to measure:

- changes in the charge-to-mass ratio distribution (mass spectrometer) which directly affect mass efficiency and thrust noise.
- changes in the plasma potential distribution (emissive probe) which reveal changes in the thruster-neutralizer interaction. When used in conjunction with the PIC simulation, this is a higher fidelity diagnostic of the neutralizer effectiveness than gross changes in the neutralizer current-voltage characteristics.
- changes in the current density distribution (Faraday probe rake) which directly affect changes in the thrust vector centroid over time.
- physical changes to components and surfaces (in-situ optical camera) resulting from erosion or vapor deposition. This will expose potential failure modes due to sputter erosion of the accelerator or coating of insulators which may not be evident except in a long duration test.

3.2.6 Micronewton Thruster Development Budget and Schedule

The schedule for the micronewton thruster development, starting in 1999, is given in Figure 3.6. The budget required to perform the development not presently funded is given in Table 3.4.

Table 3.4 Micronewton thruster development budget

Task / FY funds ($k)	1999	2000	2001	2002	2003	2004	2005	Total
Requirements definition		10						10
Performance characterization tests		210	90					300
Analysis/modeling		200	170	100				470
Lifetime tests			40	290	145	130	75	680
Total		**420**	**300**	**390**	**145**	**130**	**75**	**1460**

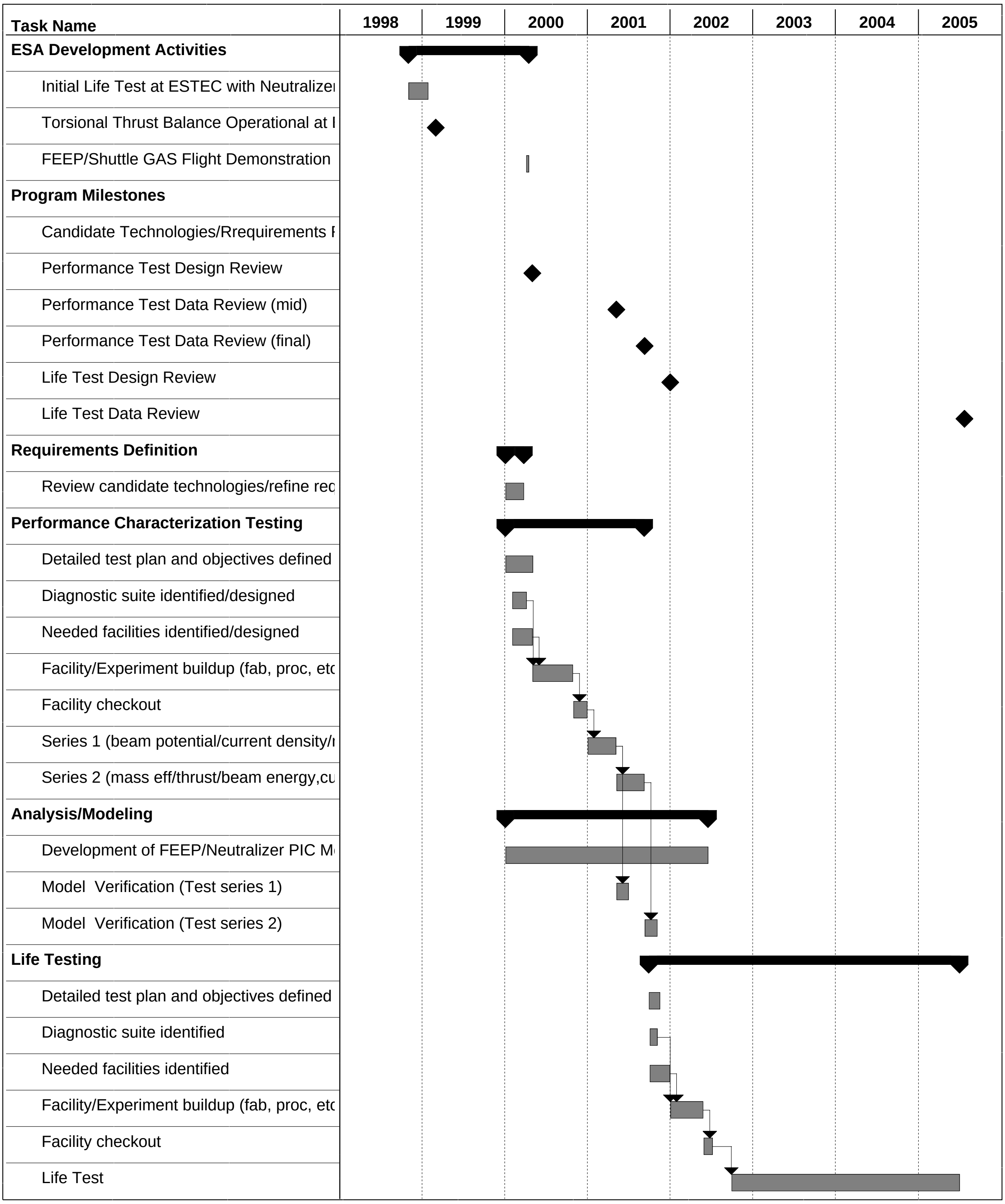

Figure 3.6 Micronewton thruster schedule and milestones

3.3 Picometer Interferometry

3.3.1 Interferometry Noise Sources

The major contributors to the LISA interferometry noise are presented in Table 3.5, which is taken from the LISA Pre-Phase-A Report (Bender et al. 1998). The technology development program will verify the noise sources individually, using a combination of bread-board experiments and modeling. The experimental program will feed into the interferometer design, both by revealing unanticipated phenomena, and by providing experience on the best way to build LISA's sub-systems.

Table 3.5 Major sources of optical path noise, and schemes to suppress their effects. The noise sources are expressed in equivalent displacement, with the "Error" column in units of pm/ Hz. For example, the pointing error effect is a combination of wave front errors, static, and fluctuating pointing errors; the budget allocates 10 pm/ Hz to each of 4 telescopes when these offsets and fluctuations act in combination.

Error Source	Error	Number	Error Reduction Approach
Detector shot noise 1 W laser; 30 cm optics	11	4	Optimize efficiency of optical chain
Master clock noise	10	1	Ultra-stable oscillators and stabilization procedure
Residual laser phase noise after correction	10	1	
Laser beam-pointing instability	10	4	Active stabilization of angular orientation of proof masses and spacecraft
Laser phase measurement and offset lock	5	4	Low noise electro-optic design
Scattered-light effects	5	4	
Other substantial effects	3	4	Careful mechanical and optical design
Total path difference	40		= measurement error of round-trip light time change

The shot noise error is expected to be straightforward to verify. The low detected optical power (approximately 50 pW) means that a detector system with a noise level of 10^{-5} radian/ Hz is adequate; this is to be compared with the sensitivity of 10^{-10} radian/ Hz that was demonstrated in ground-based detectors operating at the shot noise limit with much higher power (refer to MIT "phase noise interferometer" paper), albeit at higher frequencies. The scheme of interfering the weak detected signal with a strong local oscillator eliminates the need for photon-counting detectors, such as avalanche photodiodes. Standard InGaAs photodiodes should be adequate. The experiments to verify shot noise performance are naturally combined with the phase noise experiments (see section 3.3.3.1).

The master clock noise budget entry refers to noise on the ultra-stable oscillator that is used as the time base for phase measurements of the interferometer beat signals. Flight-qualified oscillators have phase noise at 1 mHz that is a factor of 10^4 higher than allowed by the error budget. Therefore the oscillator will be stabilized by modulating the USO frequency on the laser signal and using the separation between spacecraft as a phase reference. The experimental effort will test this concept by imposing USO modulation on beams in the laboratory and using the side band information to stabilize a USO.

Reduction of laser phase noise will be verified by locking a high-power (~1W) local laser to a low-power (~50 pW) beam. Early experiments along these lines have demonstrated the technique (see section 3.3.2.3).

The far-field effect of telescope pointing error on optical phase is (LISA Pre-Phase A Report) is

$$\phi(f) = 6\times10^{-5} rad/\sqrt{Hz}\left(\frac{d}{\lambda/10}\right)\left(\frac{\theta_{DC}}{20nrad}\right)\left(\frac{\theta(f)}{7nrad/\sqrt{Hz}}\right)$$

where $\phi(f)$ is the optical phase noise, d is the average surface error of the telescope, θ_{DC} is the static pointing error, and $\theta(f)$ is the noise in the pointing. Each of the terms in brackets relates to a separate technology development: surface quality of optics, offsets in the pointing system, and effective loop gain of the pointing system. The laser beam-pointing stability will be tested by subjecting the optical bench to a controlled temperature change and monitoring the resulting angular changes of the beam, with and without active control of pointing. These measurements will be combined with thermal models that predict the temperature environment to estimate alignment effects in orbit. Most of the effort is expected to be in designing and testing a scheme for sensing and controlling the telescope pointing.

There are likely to be second-order, unanticipated, or subtly coupled noise sources that result when several subsystems are combined. To flush out such effects, in parallel with the individual subsystem tests an optical setup that includes versions of many of the full detector's subsystems will be built up. For example, scattered light effects interact with the alignment and frequency stabilization errors. A fairly complete interferometer system and a computer simulation will be built to demonstrate that scattered-light noise does not dominate the error budget. The test-bed plans described below will include provisions for varying the amount of scatter, to verify the models and performance.

3.3.2 Interferometry Sub-Systems

3.3.2.1 Laser

In order to achieve the required shot noise level, the current mission design calls for each spacecraft to contain two active high-power continuous lasers. For maximum power efficiency, Nd:YAG lasers will be used operating at a wavelength of 1.0 µm and an optical output power of 1.0 W. The lasers must be able to operate over the 3-year prime mission. The lasers must be also of narrow intrinsic line-width so that they can be stabilized in power and frequency to the required levels. Currently there is no space-qualified laser meeting these requirements.

Lasers with requirements similar to the LISA requirements are currently undergoing testing and evaluation. The SIM project is currently evaluating lasers operating at 1.3 µm wavelength with

300 mW output power and long required lifetime. The TES project is evaluating a similar laser, which has been running continuously for three years These are adaptations of commercially available lasers. The units currently under test at JPL have been developed starting from commercial versions of NPRO Nd:YAG lasers made by Lightwave Electronics. They incorporate modifications which are believed to improve the ability of the laser to perform adequately in space, after surviving launch. In order to satisfy LISA requirements, more modifications will be necessary. In preliminary discussions with vendors it appears that a space-qualifiable laser could be procured with about 1 year lead time. These lasers would then have to be subjected to an extended performance testing program. In parallel, there is an effort underway in Germany to develop and test a laser specifically for the LISA mission (Peterseim et al. 1998). There is also increasing interest among the satellite communications industry in similar lasers for communication between satellites.

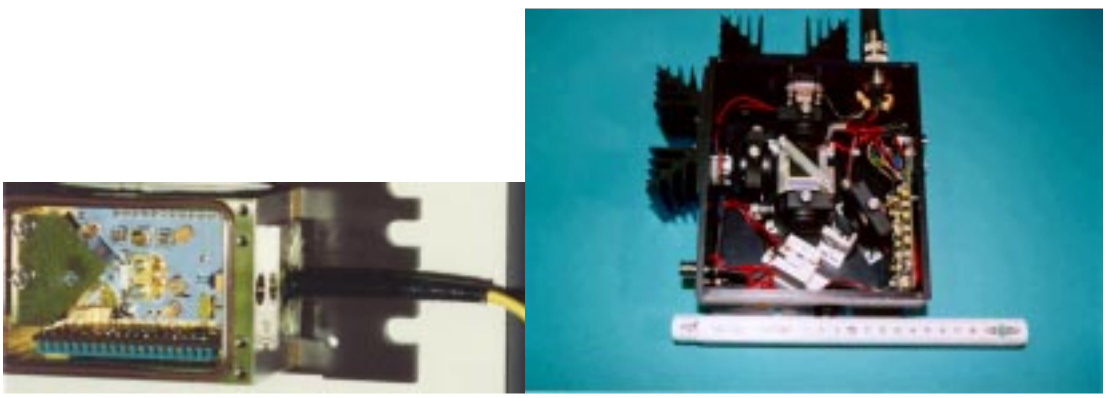

Figure 3.7 Left: Engineering laser under test by SIM project. Right; laser prototype at Laser Zentrum Hannover.

Under the proposed plan, two candidate space-qualifiable lasers will be procured and tests begun for performance and lifetime. Environmental tests will be performed, including vacuum operation and operation after launch vibration. In case of failure, the units will be repaired and re-tested. Performance tests will be done to determine that the lasers meet the power output and power and frequency stability requirements. An additional pair of lasers will be procured for the interferometer test beds. Lifetime testing, i.e. test under continuous operation, will be carried out for four years. Two years of successful operation are acceptable, since the baseline mission includes one spare laser for each optical assembly.

3.3.2.2 Laser Stabilization

Since LISA is an unequal-arm interferometer, noise in the laser frequency is not perfectly canceled between the two arms. The frequency noise is approximately canceled using an algorithm to account for the arm lengths (Giampieri et al. 1996, Tinto and Armstrong 1998). The noise cancellation algorithms require knowledge of the length of the arms of the interferometer. The amount of error that can be tolerated in the knowledge of the arm lengths depends upon the level of the laser frequency noise.

The LISA interferometry error assumes that the laser can be locked to a thermally stable reference cavity at the spacecraft operating temperature near 300K. The resultant frequency noise level is assumed to be 10 Hz/ Hz at an observation frequency of 1 mHz.

The level of frequency noise desired has been previously demonstrated in certain laboratory demonstrations but not in exactly the manner envisioned for LISA. The required frequency noise level has been demonstrated for two lasers locked to the same cavity (Salomon et al. 1988). It has been also demonstrated for two lasers locked to independent cryogenic cavities (Seel et al. 1997).

To demonstrate the baseline LISA design, two lasers will be locked to two separate room-temperature reference cavities. These reference cavities must be extremely well isolated from the laboratory environment since laboratory vibration and temperature fluctuations will cause variations in the cavity dimensions, causing frequency noise in the lasers locked to them. An experiment was begun in 1998 at the University of Hannover to fabricate two cavities in carefully isolated chamber. Lasers are locked to these cavities and the outputs compared to determine the level of frequency noise in each. The initial results of these experiments is shown in Figure 3.8 (Peterseim et al. 1998). The required LISA performance has not yet been demonstrated, but successful demonstration is expected in 1999.

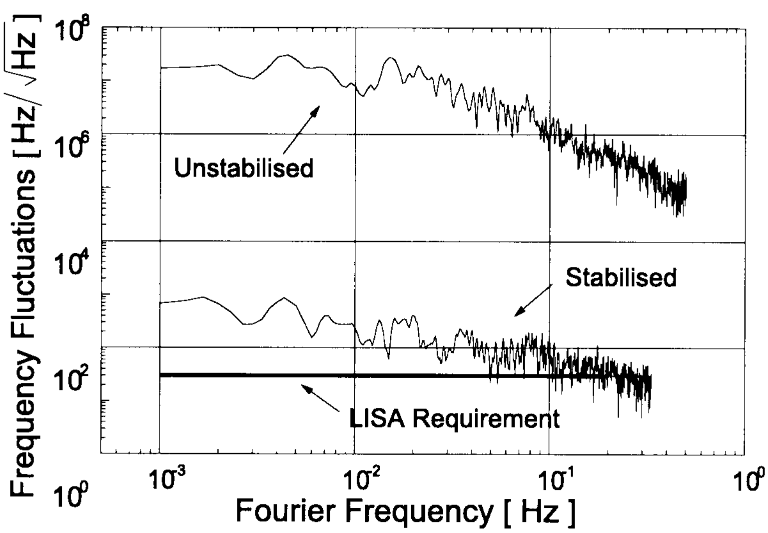

Figure 3.8 Comparison of frequency of two lasers stabilized to independent room-temperature cavities. (Peterseim et al. 1998).

3.3.2.3 Laser Phase Locking

In the simplest mode of operation of the LISA interferometry, some lasers are locked not to a local reference cavity but instead to a laser signal originating from a spacecraft 5 million kilometers away. Noise associated with the phase-locking is required to be smaller than the laser shot noise. The required level of phase-locking noise has been demonstrated in laboratory experiments (Salomon et al. 1988). However, the amount of received light for LISA is very small, only about 50 pW, much smaller than the power levels normally used for laser phase-locking. An experiment to demonstrate phase-locking with the low power levels expected for LISA has been begun at Glasgow University (McNamara et al. 1998). As shown in Figure 3.9, these have demonstrated reduction of phase noise to a level within a factor of 6 of the LISA goal for frequencies between 10^{-4} and 10^{-3} Hz, and to within a factor of 2 of the LISA goal above 10^{-3} Hz. It is expected that further work on this experiment will demonstrate the performance needed.

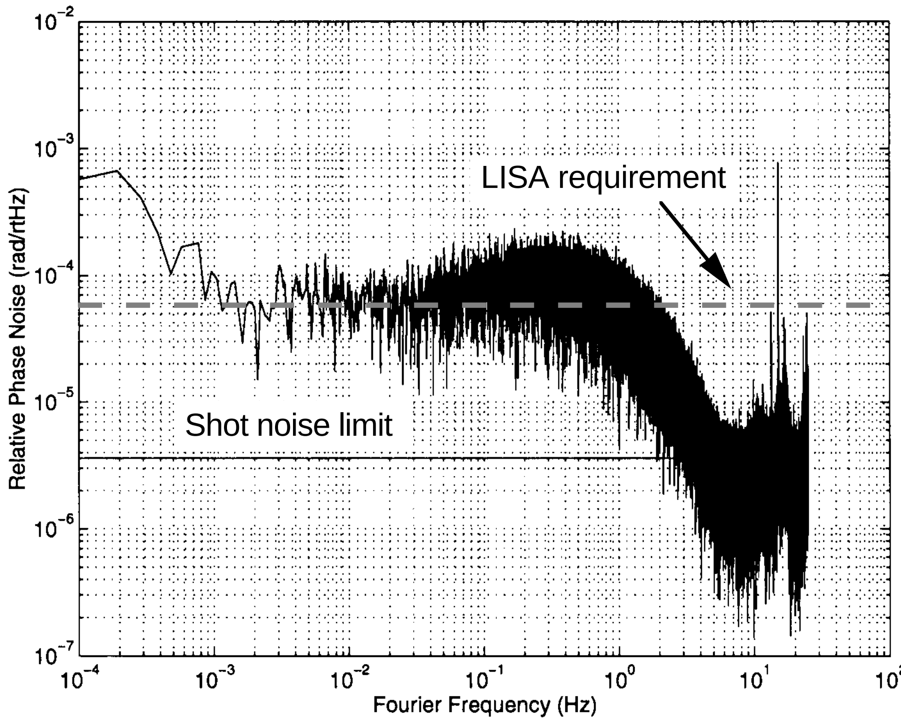

Figure 3.9 Residual phase noise in a weak-light phase locking experiment, in which two independent diode-pumped Nd:YAG non-planar ring oscillators were locked together with a 15 MHz frequency offset (McNamara et al. 1998). The dashed horizontal line represents the phase noise goal for LISA. The solid horizontal line is the shot noise level for this experiment, in which the low-power beam had a factor of 240 higher power than is planned for LISA.

3.3.2.4 Laser Electronics

In order to operate the lasers to be used in the LISA interferometry and system test beds, electronics will be needed to control the lasers and interface to the rest of the system. The components of the laser electronics, shown in Figure 3.10, are typical of a standard laser frequency and power stabilization arrangement.

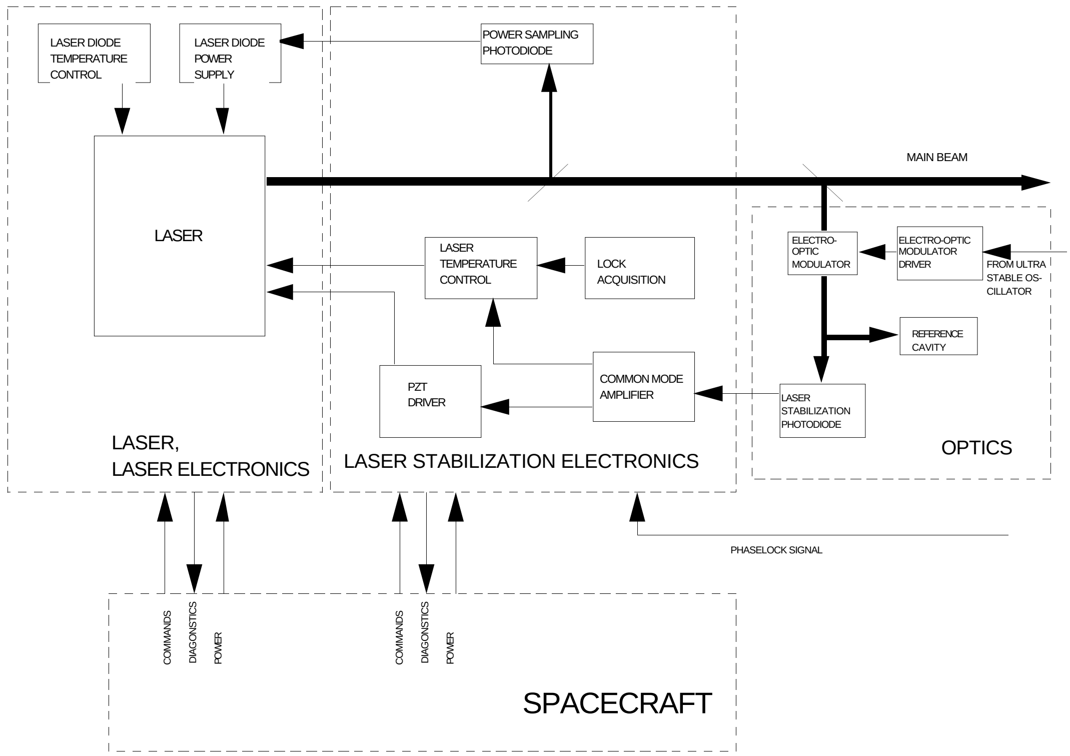

Figure 3.10 Block diagram of laser electronics

The main features of this arrangement are:

- Frequency stabilization is performed by locking the laser frequency to a resonance of a reference cavity, using a standard reflection locking technique, described in some detail in Section 3.3.2.2.
- Lock acquisition is achieved by scanning the laser crystal temperature in small steps, until the frequency of the laser is close to a resonance of the reference cavity and fringes are detected at the laser stabilization photo-detector. At this point, control is transferred to the laser frequency stabilization arrangement.
- Optical power stabilization is carried out by sampling the power in the main beam with a dedicated photodiode and feeding an appropriate correction signal via the pump laser diode power supply.
- Given the noise, bandwidth and range requirements for the frequency and power stabilization subsystems, these need to be implemented using analog electronic designs.
- While the relevant trade studies are yet to be conducted, it is likely that the design will have to make heavy use of components with low noise and low temperature dependent offset. It is possible that, in order to achieve the necessary long term frequency an power stability, the electronics will have to be housed inside an enclosure with high degree of temperature control.
- In order to enable system diagnostics from the ground, a number of parameters like pump diode laser current, optical power, various temperatures, etc., will be digitized and added to the telemetry data stream.

Breadboard versions of these electronics will be developed, with two units delivered to the Integrated System test bed.

3.3.2.5 *Optical System*

The interferometer system is based on a 30 cm telescope for transmission and reception of laser signals and an optical bench for housing the inertial sensor and various optical components, as shown in Fig. 1.5. More detail on the optical bench is shown in Fig. 3.11. In order for the interferometry system to be able to measure the distances between proof masses to the picometer level, the optical system must be sufficiently stable, both to keep the necessary component alignment during launch and in responses to changes in the science operations mission phase. The optical system must be analyzed in detail to determine the optical response to thermal and mechanical changes in the optical system. The thermal and mechanical properties of the optical system must be analyzed in terms of the expected spacecraft environment to show that the induced optical changes are within tolerances. A means of constructing the optical system to meet the mechanical requirements, including launch vibration characteristics, is needed.

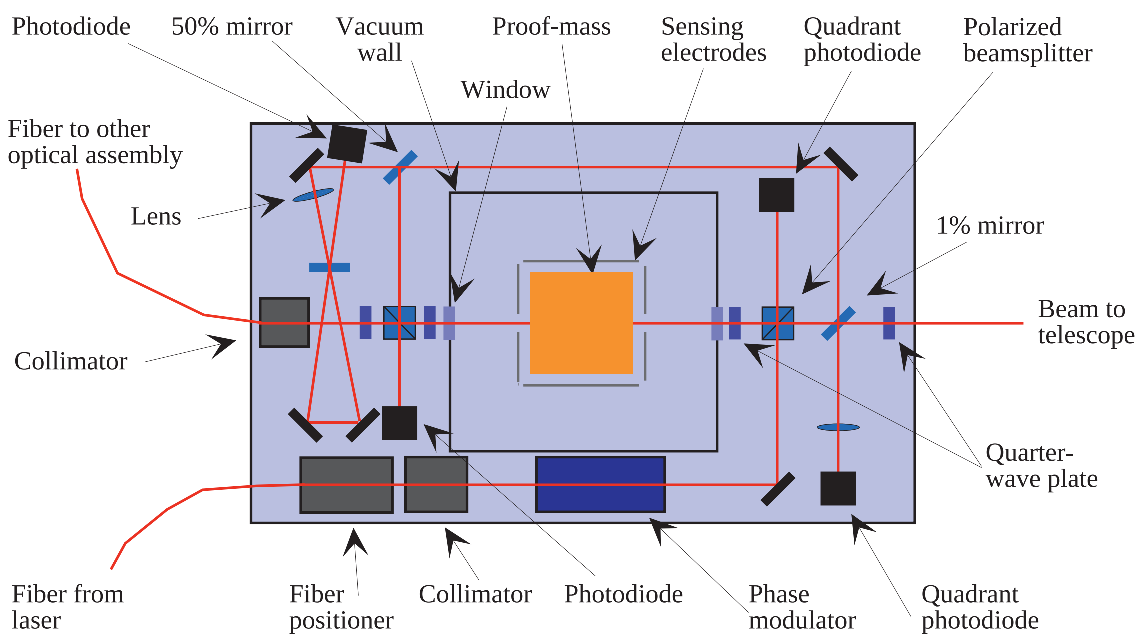

Figure 3.11 Optical bench for LISA instrument

A preliminary assessment of the LISA optical system has been performed (Caldwell et al. 1998). The optical path stability appears to be achievable with current space-qualified telescope designs. In particular, the telescope designed and constructed for the SILEX mission is about the same size as the LISA telescope and appears to meet the LISA requirements in terms of thermal and mechanical stability. A more detailed optical design is needed, to take into account the components on the optical bench and the nominal design of the mounting of the optical bench and telescope.

There are many components to be mounted on the optical bench. The optical bench itself must be made of a material with very low thermal expansion coefficient, such as ultra-low expansion (ULE) or ZERODUR glass-ceramic. Many of the optical elements are glass and could, in principle, be mounted by optical contacting, to minimize the mass of the component mounting.

Experience with GP-B has been that optical contacting is not as strong as desired, and does not allow for adjustment of the alignment of components except through expensive machining and polishing. Instead, a technique has been developed using a solvent to dissolve the glass, allowing some time to perform component alignment before the material re-solidifies into a firm flight-qualifiable bond. This technique could also be used by LISA. For GP-B this technique has been used with a quartz optical bench. Preliminary attempts to perform similar mounting using low-expansion glass have been less successful. However, it is expected that further experiments with different solvents and finishing treatments will lead to a technique that will work for LISA.

For non-glass components on the optical bench, adequate mechanical mounting techniques will be developed. Some of the mechanical mounting techniques from SIM may be used, though many will have to be adapted to the specific LISA components.

The development timeline and budget includes the development of a suitable optical bench design, development of construction and mounting techniques, and construction of two flight-qualifiable optical benches to be used in the Integrated System test bed.

3.3.2.6 Optical Components

There are a number of types of optical components needed to be mounted to the optical bench, as indicated in Figure 3.11. These include passive components, such as mirrors, lenses, quarter-wave-plates, etc., and components that need associated electronics, such as photodiodes (amplifiers), the optical modulator (high-voltage drive electronics), and fiber positioner (control electronics).

For each of these components, commercial units are available. Commercial components will need to be evaluated for operation in the space environment. Where possible, components from other missions such as SIM will be adopted to reduce cost. Others will have to be space-qualified, and some custom development may be needed.

For the associated electronics, breadboard units will need to be constructed with appropriate interfaces for use in the Integrated System test bed for evaluation and software development.

3.3.2.7 Phase Measurement Electronics

The interferometer shot-noise limit, under the baseline design, is 10 pm/ Hz. To be able to combine the signals from the various points in the interferometer system and combine them to produce the equivalent of a Michelson interferometer, a phase measurement system is needed to compare the frequencies of laser signals. For example, at the front of each optical bench, light from the local laser is mixed with light from the laser on a distant spacecraft. The beat frequencies of the various signals will be between 0 and 15 MHz based on the nominal orbit design. Higher frequencies, up to 200 MHz may be imposed on the laser signals for time-transfer information.

For the primary signals, the phase measurement system needs to provide measurements with an accuracy of better than 10 pm/ Hz, or better than 10^{-5} cycles/ Hz. The accuracy is similar to that for the SIM mission. However the dynamic range of the beat frequencies is much higher than for the SIM mission, so the SIM measurement electronics cannot be used directly.

The signal processing needed for LISA is similar to that needed for the GRACE mission. For GRACE, each spacecraft will transmit a 30 GHz signal to the other spacecraft, based on the frequency of an ultrastable oscillator, and at each spacecraft the frequency of the incoming signal is measured with respect to the local signal. For GRACE, the phase measurement system is derived from a specialized receiver developed for precision Global Positioning System measurements. The GRACE requirements are for an accuracy of 10^{-4} cycles/ Hz. This accuracy has been demonstrated in the GRACE prototype instrument. The receiver is thought to be capable of reaching better accuracy than the rest of the GRACE instrument.

The baseline LISA phase measurement system is to use the GRACE receiver. This will provide an inexpensive and flexible space-qualified capability. Early analysis and testing of the GRACE receiver is needed to demonstrate that it will reach the LISA sensitivity. The GRACE receiver software will have to be modified to accommodate the larger signal-to-noise ratio of LISA. Three receiver units will be required for operation of the Three-Arm interferometer test bed and Integrated System test bed. These can be used as engineering units for Phase C/D.

3.3.2.8 Pointing Control

Errors in the interferometry can be caused by changes in pointing of the optical system. This is because the transmitted wave fronts are not perfectly spherical. For an optical figure quality of $\lambda/20$, the system pointing accuracy is expressed as a product of the pointing bias error times the pointing jitter. To keep the path length error due to pointing error smaller than the shot noise, nominally the pointing error will be kept less than 10 nanoradian and the pointing jitter less than 10 nanoradian/ Hz.

The pointing jitter can be easily read from comparison of the phase measurements of the local laser compared with the distant laser. If the beam diameter is 10 mm on the photodiode, then a tilt between the two wave fronts of 10 nanoradian will lead to a phase difference of 5×10^{-5} cycles across the detector, which can be resolved by the phase measurement system. In fact the telescope will give an angular magnification so errors in pointing the optical instrument as a whole will be magnified by 30 to give a very easily read signal by the phase meter. Quadrant photodiodes will be used to allow determination of both pointing angles. The pointing bias will be determined by modulation of the instrument pointing and finding a point where the local and incoming beams are parallel.

Continual changes in the instrument pointing are required due to orbital perturbations on the relative positions of the spacecraft. Over the course of one year, the angle needs to be adjusted over ±0.5°. This adjustment needs to be done while keeping pointed at the distant spacecraft with 10 nanoradian accuracy.

The baseline plan calls for each LISA instrument to be hinged at the non-telescope (aft) end, with the telescope (front) end adjusted by means of a linear actuator. With dimensions of order 1 m, the actuator needs to have a precision of order 10 nm with a dynamic range of order 1 cm. The accuracy does not have to be according to a fixed response function, since feedback information from the phase measurement system can be used to correct actuation errors.

The nominal pointing actuator is expected to be a combination of piezoelectric crystal and mechanically-driven worm screw. An alternative technology is a voice-coil. Commercial devices of these types are available, and similar devices have been used in other space missions. The development plan is to identify and evaluate candidate devices for operation in the space environment. Selected devices will be tested in the single-arm interferometer test bed to demonstrate acceptable performance. Following this, candidate devices will undergo environmental tests (assuming a flight-qualified device is not identified as suitable). Units will be procured for use in the Integrated System test bed, and breadboard drive electronics developed.

3.3.3 Interferometer Test-Beds

To demonstrate the interferometry performance required for LISA, a set of test beds will be implemented. The current plan is to develop these at JPL where advantage can be taken of some of the facilities and developments done for other space interferometry missions and avoid duplication of effort, concentrating only on the innovations needed for LISA.

The single-arm test bed will be used to demonstrate the performance of the various interferometer sub-systems. There will be some overlap in tests and performance of the single-arm test bed with laboratory tests done at other partner institutions. This is because the basic principles of the interferometer are the same for wide variety of tests. The work on the single-arm test bed will concentrate on technology which is not being demonstrated at other institutions. The single-arm interferometer will also be a development stage towards the three-arm interferometer.

The three-arm interferometer will implement a triangular configuration similar to the LISA spacecraft configuration. The three-arm test bed will be used to demonstrate the operation of the interferometry system for the LISA constellation with an accuracy and frequency range compatible with the LISA mission requirements.

After completion of the three-arm interferometer goals, the optical system at one vertex will be replaced with a full scale model of the instrument system of one LISA spacecraft for integrated system testing of hardware and software.

3.3.3.1 Rigid Interferometer Test Bed

The goals of the Rigid Interferometer test bed is to demonstrate the performance of the phase measurement subsystem to the required accuracy and to demonstrate a system performance for measurement stability to the required accuracy of 10 pm/ Hz at low frequencies. These goals are consistent with the measurement requirements needed for a flight demonstration of the inertial sensors. Figure 3.12 shows the schematic layout of the interferometer (Robertson and Folkner 1998). The test bed will fit on a single optical table. An optical bench of low-expansion material, possibly super-invar or ultra-low expansion glass, will be used to mount the optical components and fixed mirrors representing the inertial-sensor proof masses. The optical bench will be placed in a thermally stabilized environment. The system will be operated to show stability of the optical paths to the accuracy of the phase measurement system to 10 pm/ Hz. The frequency response will be as close to the LISA frequency range as practical given the time and funding constraints.

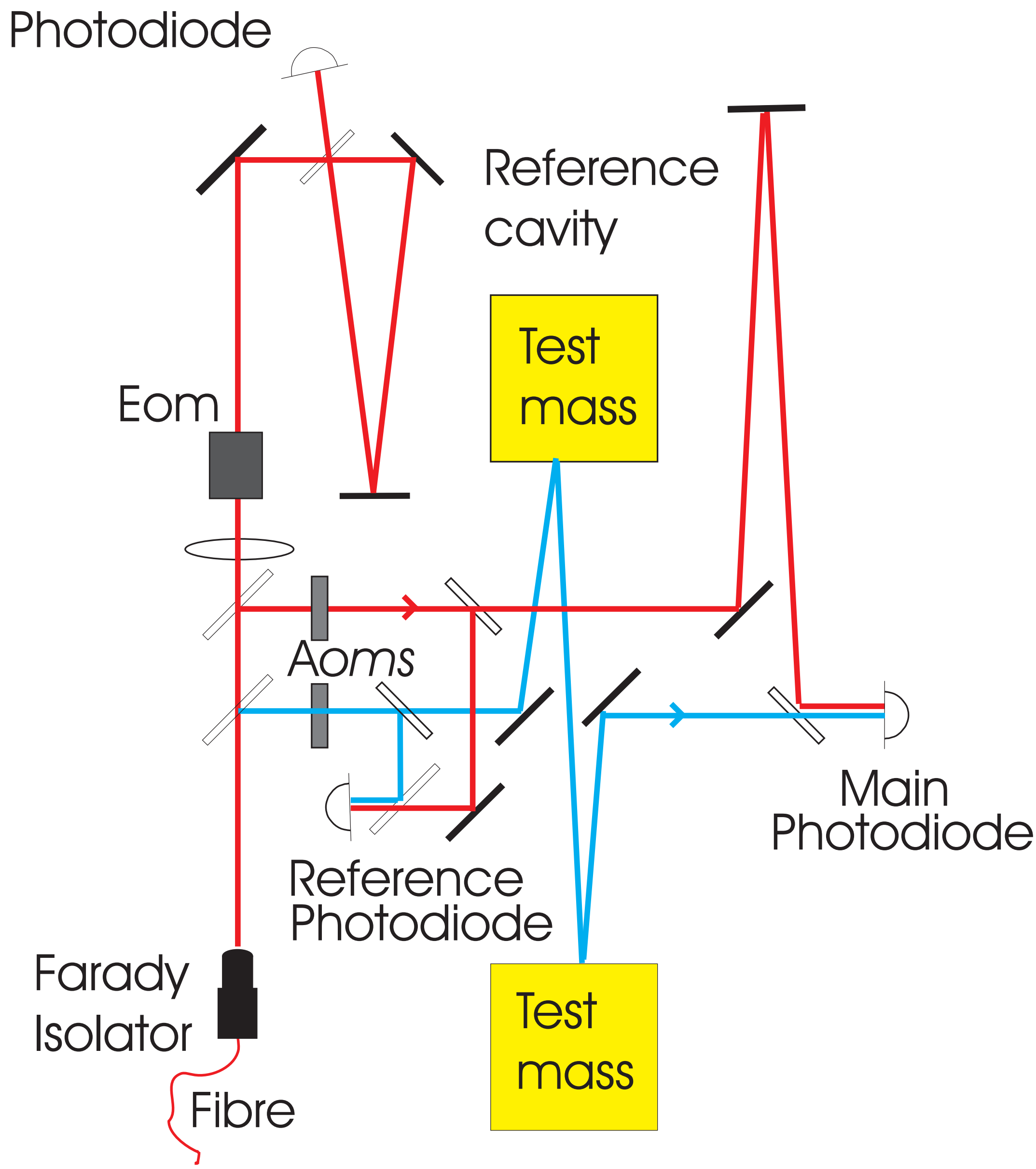

Figure 3.12 Interferometer schematic for test of inertial sensors

3.3.3.2 Single-Arm Interferometer Test Bed

The Single-Arm test bed will be used to demonstrate the performance of the pointing actuator subsystem, investigate signal acquisition techniques, and be used as a test bed for development of control electronics. Two independent optical benches will be mounted on pivots. The ends will be supported by actuators selected for the LISA instrument pointing system. Each optical bench will include laser transmitting optics and a quadrant photodiode for measuring the angle of an incoming beam with respect to the transmitted beam. Separate sensors will be located external to the optical benches to validate the information from the pointing readout. The pointing actuators will be mounted on adjustable mounts so that their performance can be evaluated over their full range.

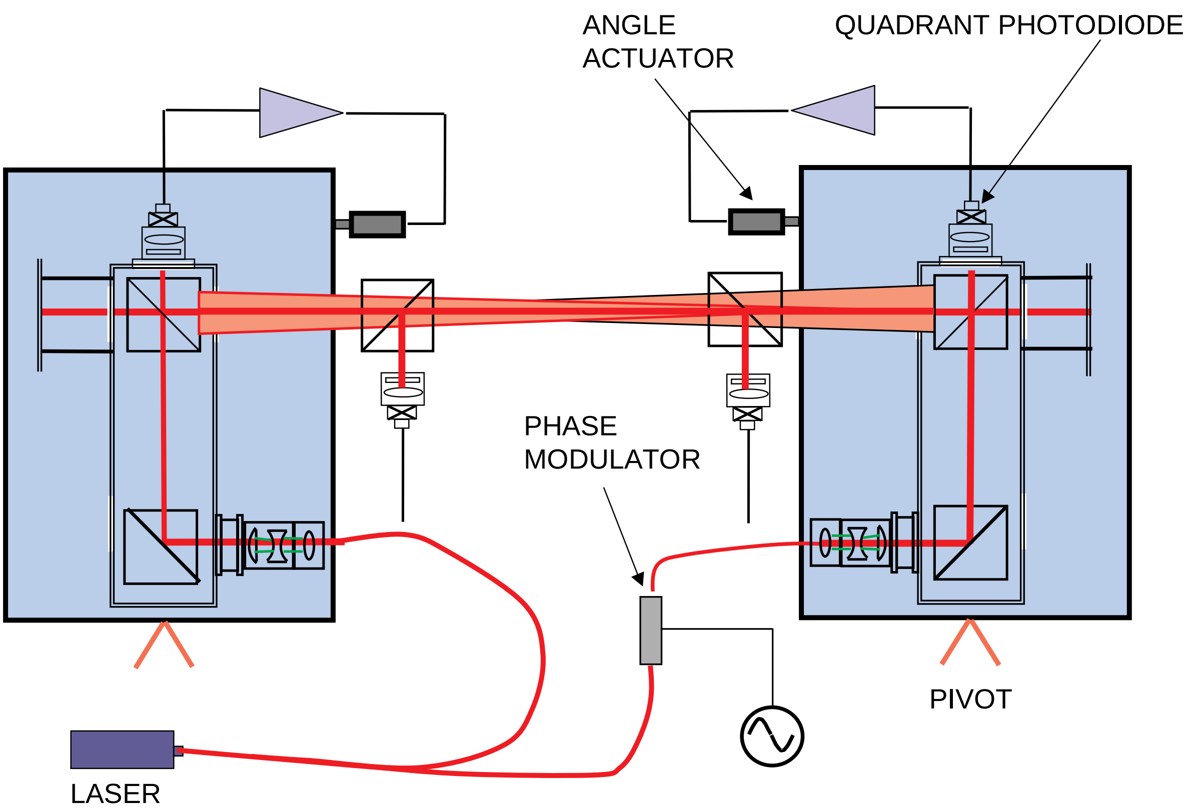

Figure 3.13 Single-Arm Interferometer test bed

3.3.3.3 Three-arm Test Bed

The Single-Arm test bed will be evolved into a Three-Arm test bed as indicated in Figure 3.14. Each vertex will be on a separate optical table. The interferometer arm lengths will be of order 5 m long. This size is determined by the desire to eventually replace the equipment at one vertex with a full-scale engineering model of a LISA payload.

Each vertex of the three-arm test bed interferometer will contain a laser and phase measurement system for comparison of the incoming and outgoing laser signals. The phase measurements will be combined in a manner similar to LISA, but in clock-wise and counter-clockwise paths to form an optical gyro, rather than difference between arms to form a Michelson interferometer. Because each path will be traversed twice, in opposite directions, there will be large cancellation of disturbances. With the small area enclosed by the three-arm interferometer, the rotational response will be small, and it will be possible to demonstrate that the system is functioning with accuracy corresponding to measuring the individual arms lengths to the 10 pm/ Hz goal for the LISA mission. The frequency response is not as certain, since it will depend to some extent on environmental noise that cannot yet be assessed. The goal is to demonstrate performance down to 10^{-3} Hz or lower.

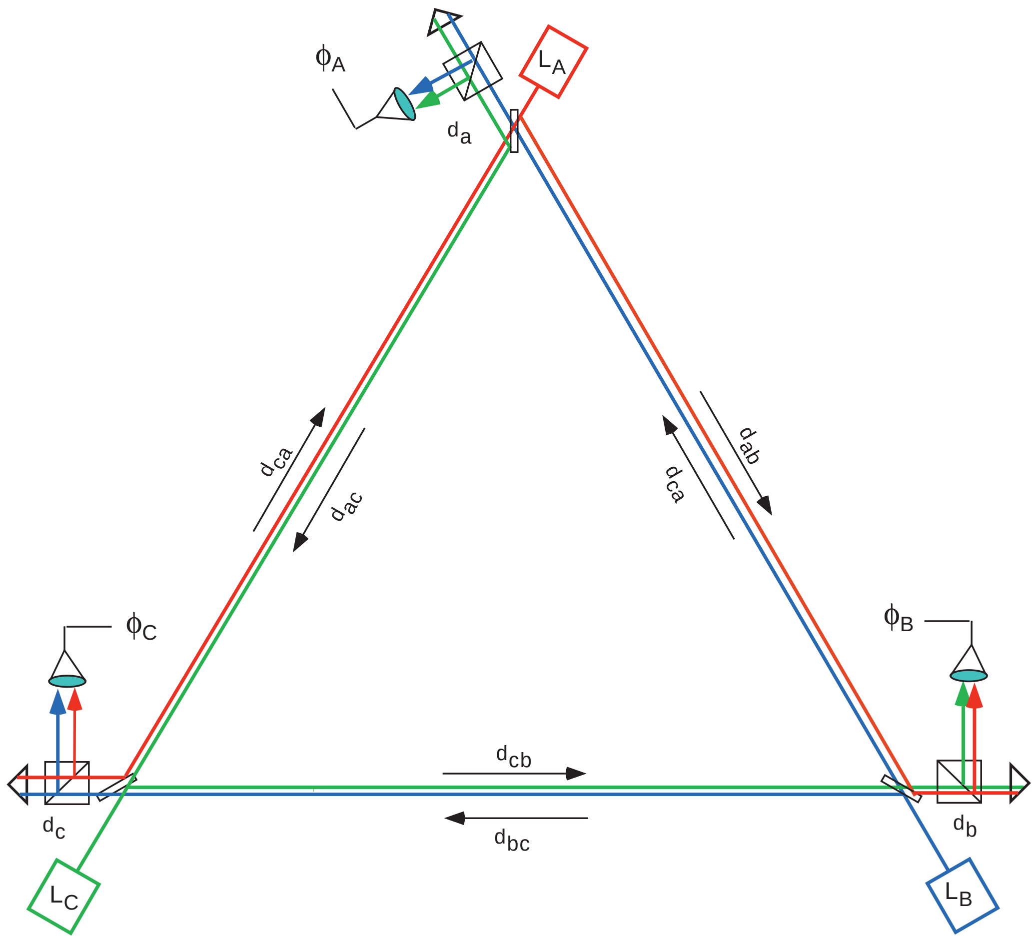

Figure 3.14 Interferometer schematic for three-arm test bed

To understand the gyro operation, assume that each laser transmits with $\phi = \omega t$ (with phase measured at the center of the primary beam splitters). (In practice the lasers would be offset slightly in frequency). At each diode beat signals are formed that are the differences in laser phases, with offsets determined by the path lengths traveled. For example, ϕ_c is the difference in the laser signal arriving from laser B and the laser arriving from laser A; $\phi_c = [\omega(t - (d_c\text{-}d_{bc})/c)] - [\omega(t - (d_c\text{-}d_{ac})/c)]$, where d_{bc} is the distance traveled from the primary beam splitter from laser B to the primary beam splitter at laser C, d_c is the distance from the primary beam splitter near laser C to the photodiode via the corner reflector and the secondary beam splitter, and c is the speed of light. Taking the sum of the phases gives the result

$$\phi_a + \phi_b + \phi_c + = [d_{ab} + d_{bc} + d_{ca} - d_{ba} - d_{ac} - d_{cb}]\ \omega /c$$

which is the difference of the clockwise and counterclockwise optical paths that form the gyro measurement. The cancellation of the distances from the beam splitters to the photodetectors and the commonality of the geometric path lengths give cancellation that will lead to improved low-frequency sensitivity.

3.3.4 Picometer Interferometry Schedule and Budget

The schedule for the picometer interferometry development is given in Figure 3.15. The milestones are shown in Figure 3.16. The budget required to perform the development is given in Table 3.6.

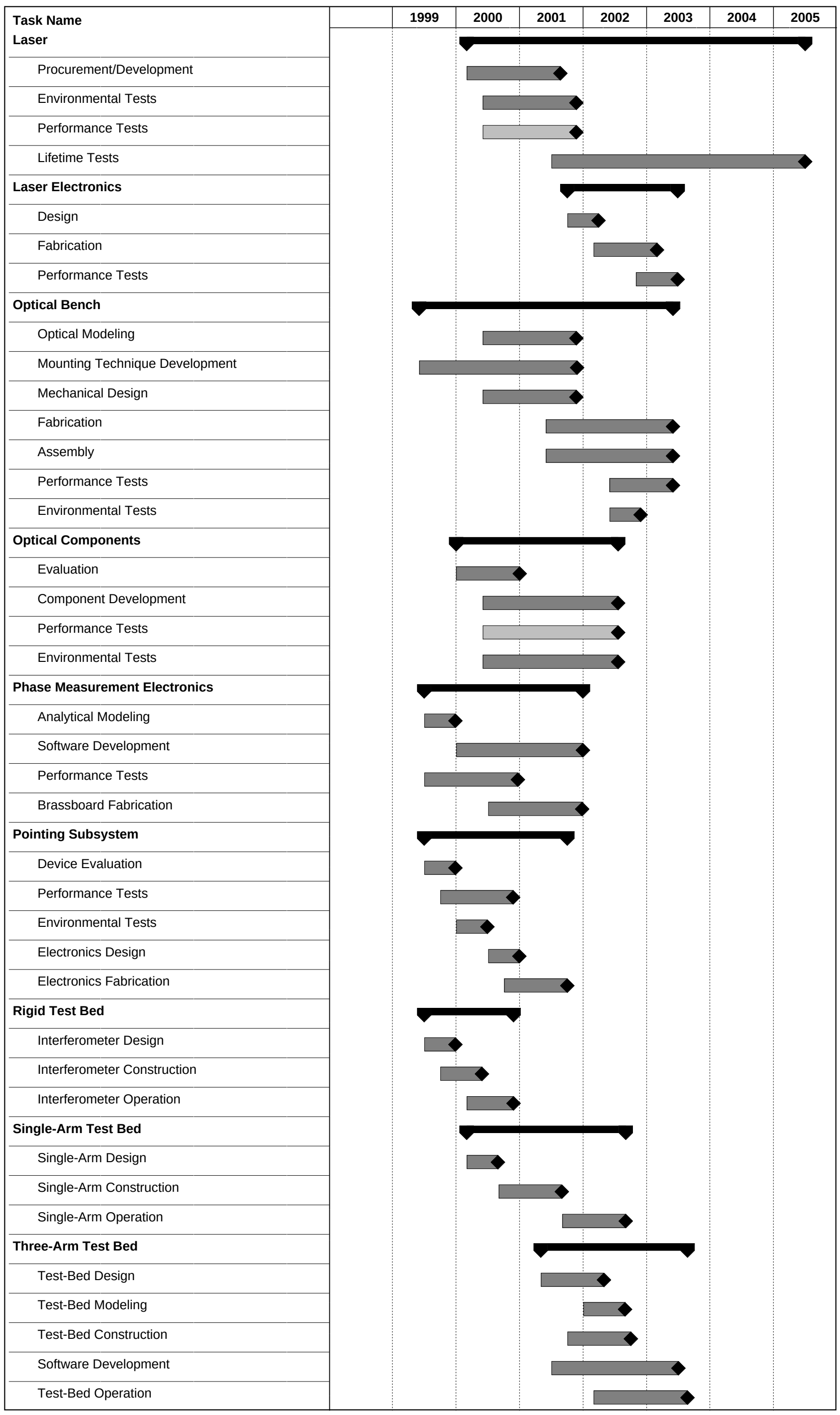

Figure 3.15 Picometer Interferometry development schedule

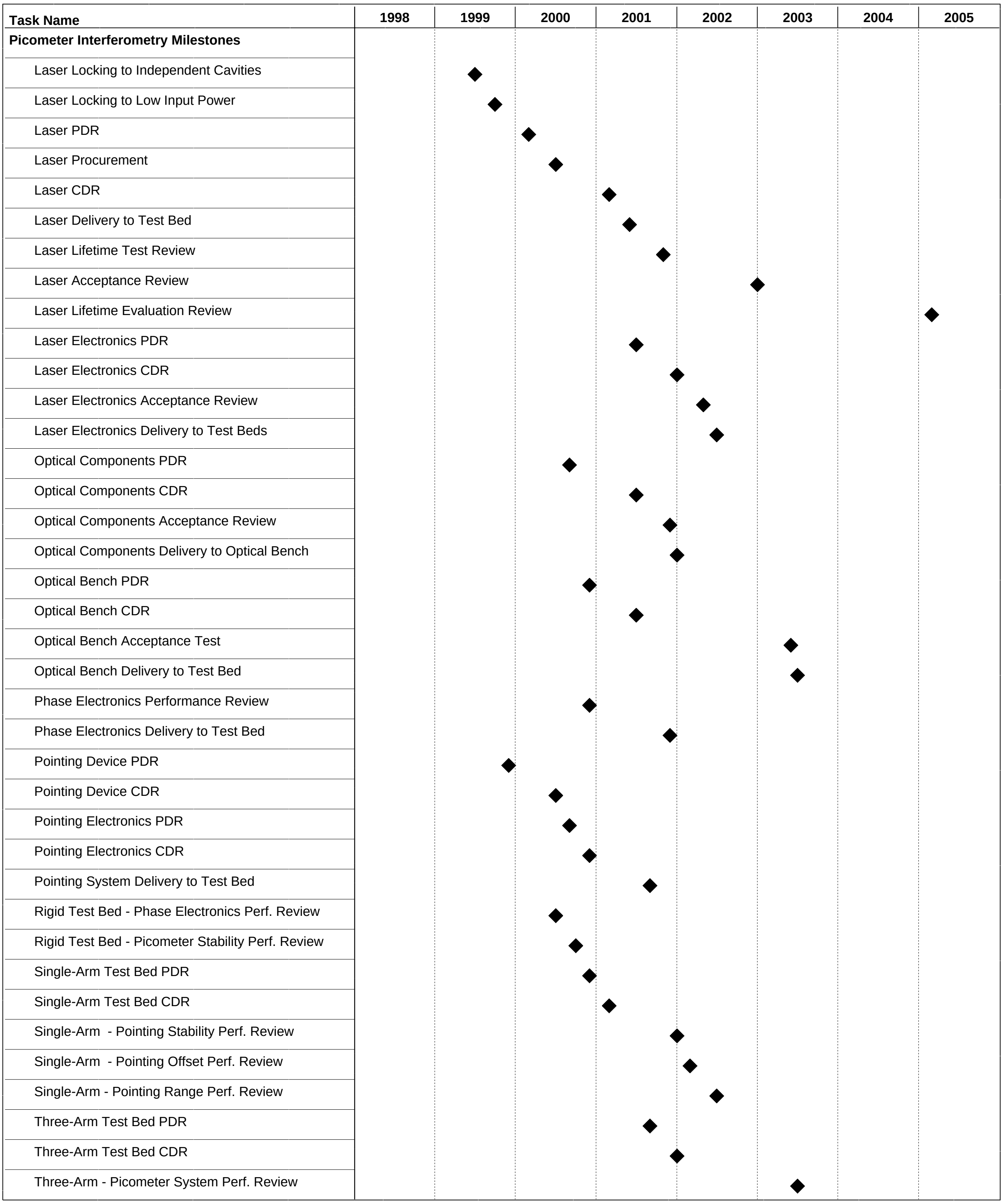

Figure 3.16 Picometer Interferometry development milestones

Table 3.6 Picometer Interferometry development budget

Task / FY funds ($k)	1999	2000	2001	2002	2003	2004	2005	Total
Laser		**375**	**425**	**50**	**50**	**50**	**50**	**1000**
Procurement (4)		200	200					400
Environmental tests		25	25					50
Test equipment		50	50					100
Performance tests		100	100					200
Lifetime tests			50	50	50	50	50	250
Laser Stabilization			**250**	**350**	**250**			**850**
Design			150	50				200
Breadboard fabrication (6)			50	100	100			250
Reference cavity (for tests)				100				100
Test equipment			50	50				100
Performance tests				50	150			200
Optical System	**50**	**400**	**700**	**900**	**850**			**2900**
Optical modeling		150	150					300
Component mounting technology	50	100	200					350
Mechanical design		150	150					300
Brassboard fabrication (2)			100	400	400			900
Brassboard assembly			100	100	200			400
Performance tests				300	150			450
Environmental tests				100	100			200
Optical Components		**625**	**475**	**475**				**1575**
Evaluation/selection		150						150
Procurement		200	200	200				600
Development		200	200	200				600
Environmental test		75	75	75				225
Phase Measurement Electronics	**50**	**275**	**250**	**150**	**300**			**1025**
Analytical modeling	50	100						150
Software development		100	100					200
Performance test		75	150					225
Brassboard fabrication (3)				150	300			450
Pointing Subsystem	**50**	**250**	**100**					**400**
Device evaluation	20							30
Device procurement	5	25						30
Performance test	25	50						75
Environment tests		50						50
Electronics design		75						75
Breadboard fabrication (2)		50	100					150
Rigid Test Bed	**100**	**300**						**400**
Interferometer design	50							50
Interferometer construction	50	100						150
Interferometer operation		200						200
Single-Arm Test Bed		**250**	**400**	**400**				**1050**
Single-arm design		150						150
Single-arm thermal/vacuum system		100						100
Single-arm construction			100	100				200
Single-arm tests			300	300				600
Three-Arm Test Bed			**950**	**1500**	**750**			**3200**
Optical/thermal/mechanical design			150	250				400
Optical/thermal/mechanical modeling			150	150				300
Thermal/vacuum system			250	250				500
Vibration isolation			200					200
Lasers and electronics				200				200
Test equipment			100	200	100			400
Software			100	200	200			500
Three-arm tests				250	450			700
Total	**250**	**2695**	**3650**	**3725**	**1950**	**50**	**50**	**12400**

3.4 System Operation

3.4.1 Spacecraft Control System Simulator

The relationship of the fine spacecraft control in relation to the traditional attitude control and spacecraft navigation systems could be implemented in several different ways. One approach would be for the traditional spacecraft control system to be responsible for all measurement processing and calculation and execution of necessary control. Another possibility would be for the spacecraft control system to allow a separate instrument process to generate control signals, while monitoring the control commands to keep within specified bounds. The advantage of this latter architecture would be to lower the cost of the spacecraft control system, partly offset by a more complex interface. The final division of responsibilities and interface definition will not be selected until Phase C/D when the division of responsibilities between ESA and NASA are finalized, and contracts are negotiated with the industrial and university contractors.

For the mission technology development phase, the baseline plan is to construct as accurate a model of the instrument as possible and develop working prototype software and spacecraft interfaces. Since aspects of the spacecraft operation cannot be implemented on the ground, a spacecraft simulator will be necessary to accept input from the instrument and provide simulated spacecraft responses. This will allow the best possible investigation of the systems control aspects of the mission.

Separately, some aspects of the system control may have to be developed in connection with a flight demonstration of the inertial sensors. Such a demonstration will retire much of the risk associated with the implementation of the fine spacecraft control. Hopefully much of the modeling and flight software will be adaptable to the LISA mission.

The development of the spacecraft simulator will begin with construction of mathematical models of the control actuator responses. A detailed mechanical model of the spacecraft and instrument will be developed in software. A model of the spacecraft response to control signals will be generated. A software model of the optical system will be developed including the interface between the optical system pointing readout and spacecraft control response. A software model of the inertial sensor will next be developed, based on the inertial sensor reference design. The system model will be evaluated for stability, and optimized for location of control thrusters and interaction algorithms.

Following the modeling exercise, software modules will be developed to interface with prototype versions of the payload electronics and with the micronewton thrusters. The high-level interfaces and software control blocks are shown in Figure 3.17. The separate modules will be developed for testing with the various test beds. For example, the pointing interface software will be used with the Single-Arm test bed used to evaluate the pointing subsystem. The software developed will not be to flight standards, but will provide functional guidelines for the later development of flight software.

With the development of the spacecraft simulator model, a six-axis motion platform will be developed for incorporation into the integrated system test bed, which the model payload will be placed on. The platform will be controlled to simulate the motion of the spacecraft in response to the signals from the model instrument. This will allow investigation into unmodeled interactions of the complete system.

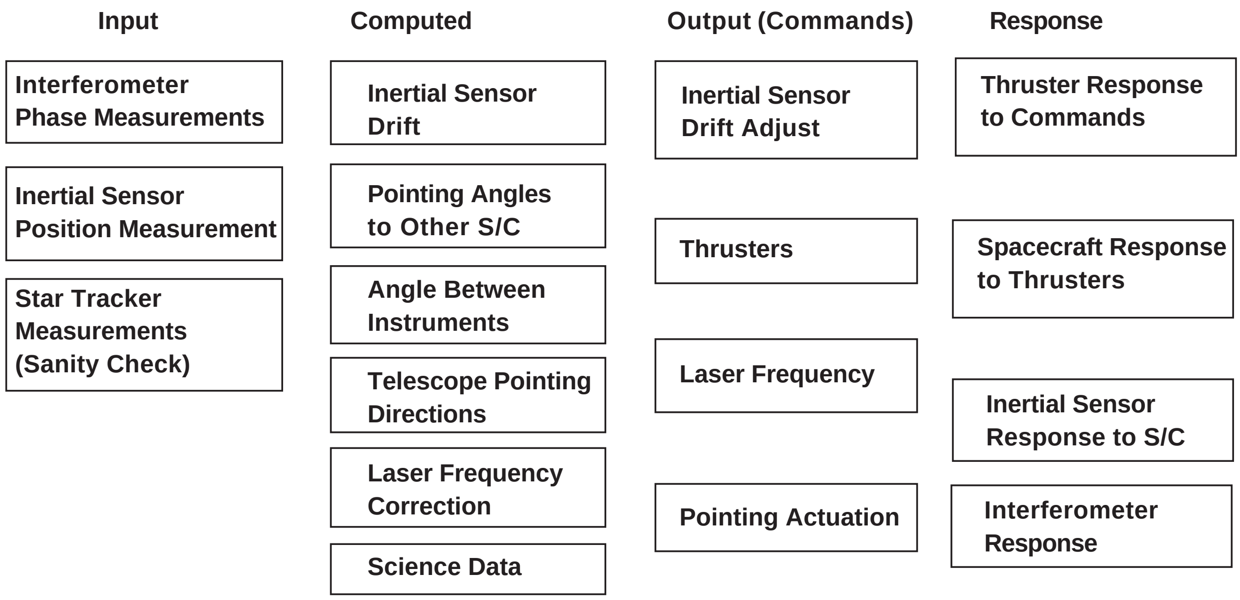

Figure 3.17 Interface and control modules for system simulation and operation.

3.4.2 Integrated System Test bed

The Integrated System test bed is designed to test an engineering model of the LISA instrument in conjunction with a simulated spacecraft. This will evolve from the Three-Arm interferometer test bed and the spacecraft control systems simulator. The complete system will be operated in conjunction with the other two vertex instruments of the Three-Arm test bed. As with the Three-Arm test bed, this will allow test of the interferometry system at the accuracy and frequency levels required, by operating as a gyro rather than a Michelson interferometer. The system can also be operated as a Michelson interferometer to check system operation, though vibration noise is expected to limit performance to higher frequencies.

The integrated system test bed will incorporate two flight-qualified optical bench brassboard units. These will be combined with an engineering model of the instrument structure built for the test bed. The instrument structure will be similar to the actual instrument structure. It need not be constructed with the composite materials needed for the flight instruments, but it might be desirable to do so, if funds are available, to retire risk associated with composite manufacturing technology. The test bed will also incorporate functional models of the LISA telescope and mount. This need not be a flight-qualifiable telescope but should be similar in construction so that interface issues can be tested.

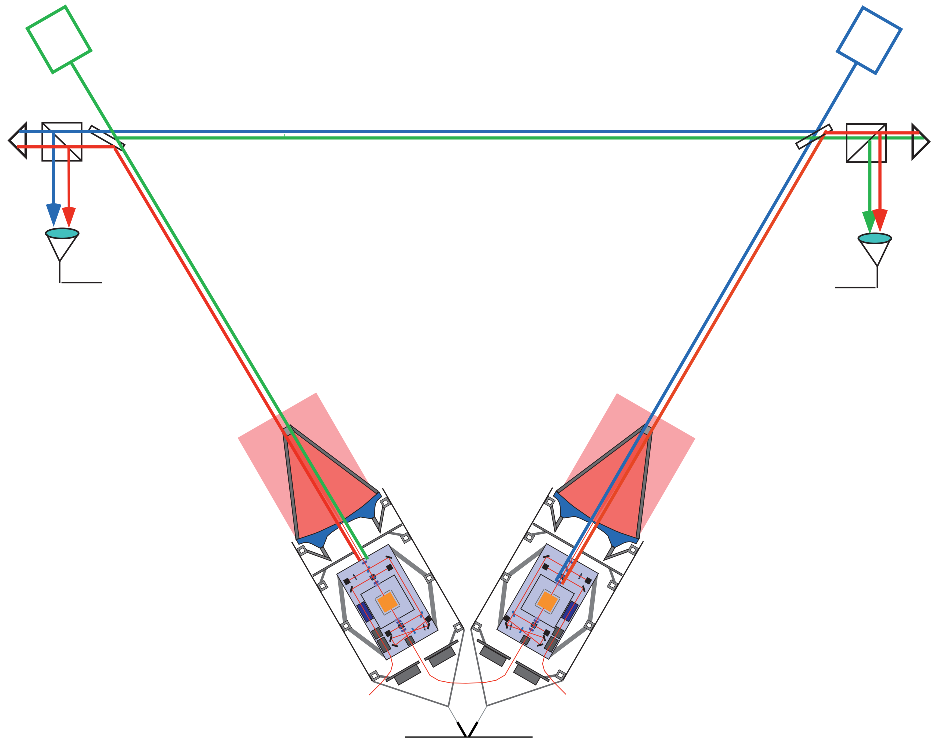

Figure 3.18 Integrated system test bed.

3.4.3 System Operation Schedule and Budget

The schedule for the system operation development is given in Figure 3.19, with milestones shown in Figure 3.20. The budget required to perform the development is given in Table 3.7.

Table 3.7 System Operation development budget

Task / FY funds ($k)	1999	2000	2001	2002	2003	2004	2005	Total
Spacecraft Simulator		**100**	**200**	**650**	**650**	**700**	**300**	**2600**
Control system design/modeling		50	100	200	200	150	75	775
Thermal/mechanical modeling		50	100	200	200	200	75	825
Software development				200	200	200	75	675
Simulator operation						150	75	225
Workstations				50	50			100
Integrated System Test Bed				**450**	**950**	**1150**	**500**	**3050**
Optical/thermal/mechanical design				300	150	150		600
Optical/thermal/mechanical modeling				150	150	150		450
Telescope design					50			50
Telescope fabrication					50	150		200
Test bed fabrication					200	300		500
Software development					150	150	100	400
System tests					200	250	400	850
Total		**100**	**200**	**1100**	**1600**	**1850**	**800**	**5650**

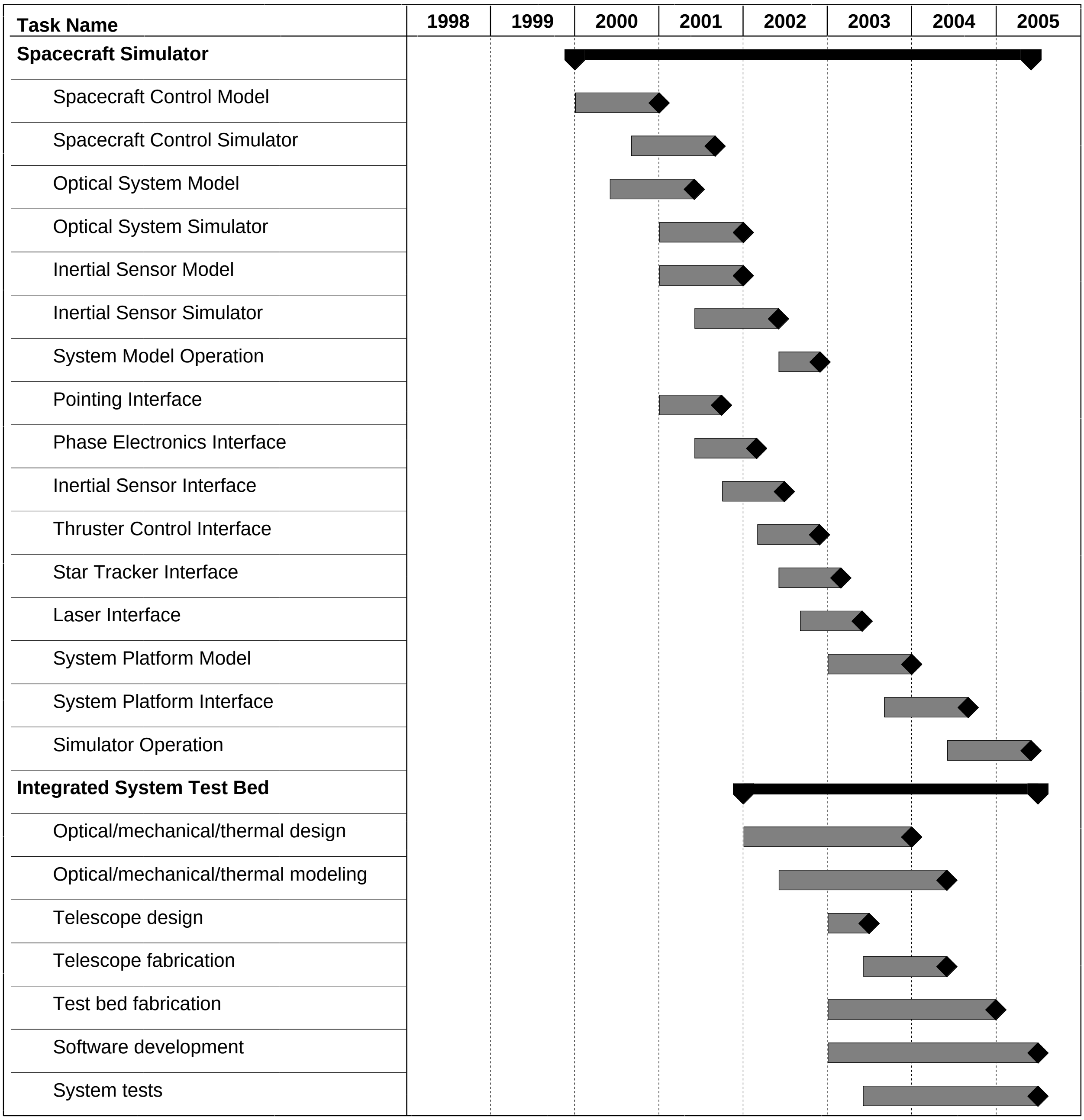

Figure 3.19 System Operation development schedule.

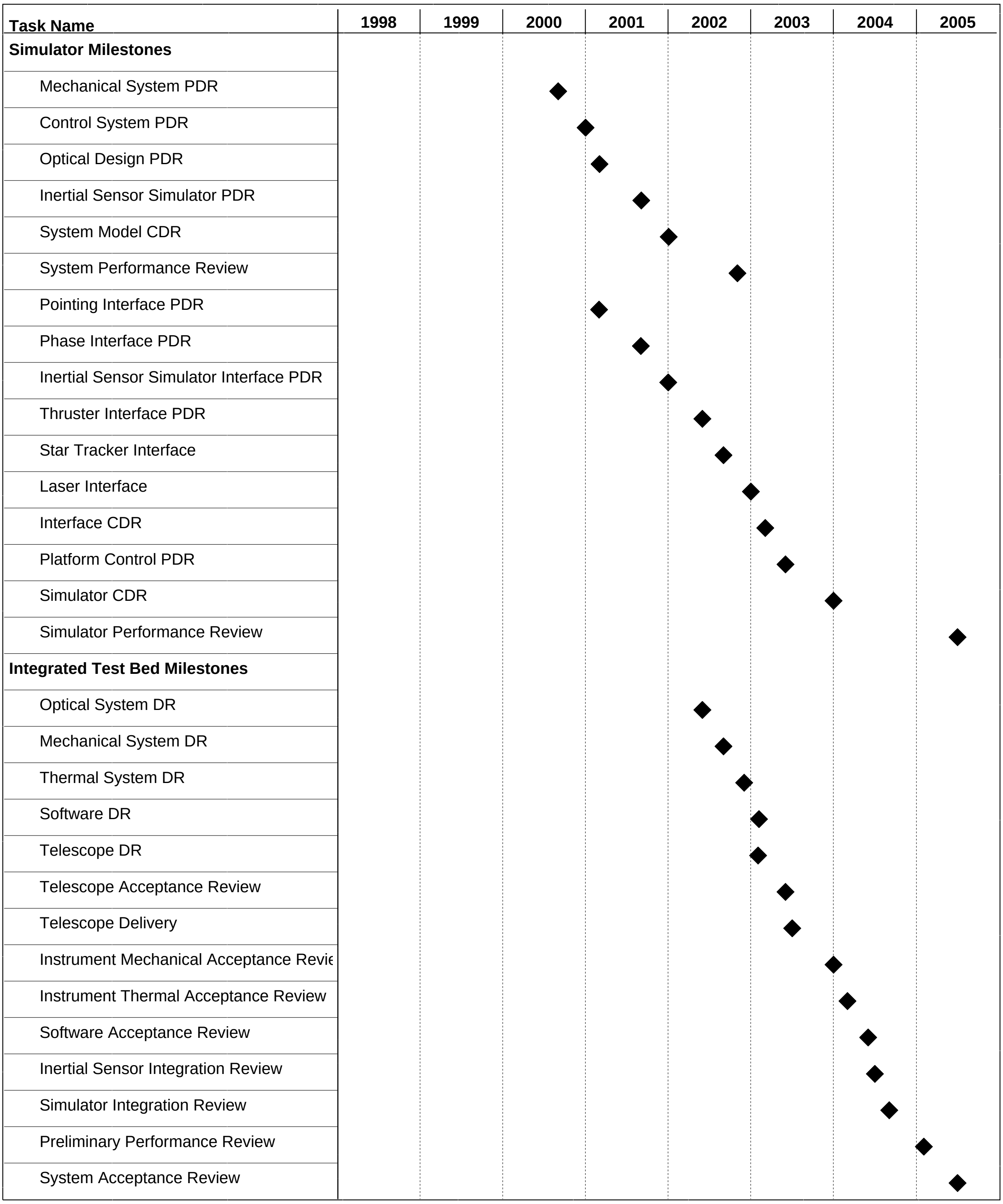

Figure 3.20 System Operation development milestones.

4. Technology Development Implementation Plan

4.1 Programmatic Assumptions

The technology development plan, schedule, and budget presented above is based on a number of assumptions. The plan is meant to show one path towards achieving technological readiness for a mission New Start in 2006 and launch in 2008. The plan assumes that an opportunity for a flight demonstration of key technologies, especially the inertial sensors, will be separately funded. Also included is a series of test beds that lead towards system-level testing of a prototype model of the LISA payload. Since funds for development for LISA-specific technology have not yet been allocated by NASA, the scale of the technology development will have to be adapted to available resources.

4.1.1 International Partnerships

LISA is projected as a shared NASA/ESA mission with participation from European national space agencies. The LISA team consists of a large group of US and European scientists working together. This cooperation is formally recognized by officially appointed representatives of ESA’s LISA Science Study Team on the US LISA Mission Definition Team, and vice versa. However no formal negotiations have yet taken place on a shared NASA/ESA mission.

ESA is funding development of the micronewton thrusters need for LISA and other missions. It is anticipated that ESA will continue this development and eventually fund the thrusters for the LISA mission. The thruster development work takes place primarily at Centrospazio, Italy, and Seibersdorf, Austria. ESA would also presumably provide significant portions of the required spacecraft systems for which technology development is not included in this plan. An ESA-sponsored industrial Phase-A study of the LISA mission is scheduled to take place in 1999.

Developments of inertial sensors have been funded in the past by ESA and by CNES. This has been principally implemented by ONERA. ONERA will be providing accelerometers for the upcoming German CHAMP mission and the NASA GRACE mission. ONERA has also recently completed a preliminary design of an inertial sensor suitable for LISA, under contract to ESA. Under CNES sponsorship ONERA is developing prototypes of inertial sensors for various tests, including perhaps a flight test at levels short of the LISA flight test goals. It is anticipated that future work in the inertial sensor area will take place in Europe. A flight test of inertial sensors for LISA would likely include candidate sensors of European origin, possibly in addition to sensors developed by the US.

Besides these specific activities by ESA and CNES, much LISA-related work has been performed by other institutions. The United Kingdom’s Rutherford Appleton Laboratory has performed preliminary optical, thermal, and mechanical design studies of the LISA payload. RAL will also lead the ESA Phase-A mission study. In addition to ONERA, the University of Trento, Italy, Imperial College, UK, and the University of Birmingham, UK have participated in studies of the performance of inertial sensors. The Max-Planck Institute for Quantum Optics and Glasgow University have been extensively involved in studies of the LISA interferometer system and verifying aspects of system performance. The Lazer-Zentrum Hannover is working on laser design for space use, including the LISA mission and precursor flight demonstrations. It is anticipated that the participation of these institutions will continue to grow and that they will be joined by other European institutions as development progresses.

Interest in the LISA project has been expressed by non-US, non-European scientists. Participation by the appropriate space agencies is not yet active but may materialize in the future.

4.1.2 University and Industry Involvement

Several universities play key roles within the LISA project. European universities that are active participants in the LISA development are listed in Section 4.1.1. Within the US, the University of Colorado has been involved in studying the overall mission concept through several design iterations. Stanford University has key expertise in inertial sensors and drag-free spacecraft operation, through the TRIAD and Gravity-Probe B projects. Because of GP-B, Stanford has techniques for mounting precision optics that will be applicable to the LISA instrument. Other universities represented on the LISA Mission Definition Team that may have active roles in the technology development for LISA include the Massachusetts Institute of Technology and the University of California at Irvine. US activities in LISA technology development have been, and are planned to continue, via contract from JPL.

Industry involvement in the technology development is yet to be defined. There are several technology developments areas which could involve substantial industry roles, such as fabrication of the optical bench, laser development, electronics development, development of major test beds.

4.1.3 NASA Center Roles

JPL is the NASA center responsible for LISA and where the LISA Pre-Project office resides. JPL is expected to be the coordinating center for LISA technology development, and lead NASA center for LISA interferometry. At JPL, LISA can take advantage of personnel and facilities taking part in other space interferometry projects, such as the Space Interferometer Mission and Terrestrial Planet Finder. JPL also has extensive experience with ion engines, of the type planned for LISA orbit injection. That expertise and related facilities can also be applied to tests of the micronewton thrusters.

Goddard Space Flight Center has an active interest in LISA. GSFC is currently working on optical modeling of the LISA instrument, taking advantage of expertise in modeling of telescopic systems for other space missions. Other roles for GSFC or other NASA centers may be identified in future.

4.1.4 Resource Assumptions

Implementation of the LISA technology plan is dependent on funding from NASA's Office of Space Science. The plan assumes that the development can begin with a small amount of funding in FY '99 with substantially more funding in future years as shown in Table 4.1.

Various facilities could be made available for LISA technology development if the assumed funding profile is followed. The major test bed activities are assumed to take place at JPL. Initially at JPL the small test interferometers can be constructed in parallel with work for the New Millennium DS-3 Separated Spacecraft Interferometer. Later, major LISA test beds could take place in the area allocated to the DS-3 test beds. Alternatively, the test beds could be in a new building being planned at JPL for major interferometry projects. 8 senior In parallel small interferometers designed to test specific aspects of the interferometry are under development at the University of Colorado. Also, there are major facilities at Stanford University, currently allocated to Gravity Probe B, which may be available for LISA related work as GP-B moves towards launch in 2001.

The LISA work force is currently small due to lack of funding. However the LISA Mission Definition Team includes 29 members, and the LISA Science Study Team includes about 20 more. In addition, there is a core group of about 10 senior personnel at JPL contributing expertise to the LISA mission on a limited basis. A similar sized group at Stanford University is involved on a limited basis, and could become available soon as GP-B moves towards launch. A few post-docs and graduate students at US and European institutions are or could soon be involved in LISA-related activities is funding were available. The work force needs to be ramped up, but the main limitation to doing so is the funding required to support them.

Table 4.1 Technology development budget

Task / FY funds ($k)	1999	2000	2001	2002	2003	2004	2005	Total
Inertial Sensors	50	450	2300	3800	2500	1100	300	10500
Micronewton thrusters		420	300	390	145	130	75	1460
Picometer Interferometry	250	2725	3650	3725	1950	50	50	12400
System Operation		100	200	1100	1600	1850	800	5650
Reserve			500	500	1000	1000	300	3300
Total	**300**	**3695**	**6950**	**9515**	**7195**	**4120**	**1525**	**33310**

4.2 Implementation Plan

4.2.1 Work Breakdown Structure

The Work Breakdown Structure (WBS) for the LISA Technology Plan. The WBS is derived from the technology tasks described in Section 3. The Integrated System Test Bed is included under Picometer Interferometry within the WBS since it is closely coupled to the Three-Arm Test Bed. The Spacecraft Simulator is broken out separately since it is more amenable to being performed as a unit independent of the main flow of test beds. As funding becomes available, major sub-tasks of the WBS are expected to be contracted out, and the WBS modified to reflect the structure of the contracts.

4.2.2 Requirements Definition

The currently envisioned requirements for LISA technology development are given in Section 3, mainly derived from the LISA Pre-Phase A Report. The Requirements Definition element of the Work Breakdown Structure will develop specific system and subsystem requirements for each element of the WBS traceable to the LISA project requirements. The Technology Working Group will be formed to provide advice and external reviews of the requirements.

Table 4.2 Work breakdown structure

1 Technology Development Management
- 1.1 Management Staff
- 1.2 International coordination
- 1.3 Reviews
- 1.4 Electronic Data System
- 1.5 Reserve

2 Requirements Definition
- 2.1 Technology Working Group

3 Inertial Sensor
- 3.1 Inertial Sensor Management
- 3.2 Design Development
  - 3.2.1 Design Trade Studies
  - 3.2.2 Performance Modeling
  - 3.2.3 Detailed Design
- 3.3 Prototype Development
  - 3.3.1 Proof Mass Fabrication
  - 3.3.2 Reference Housing
  - 3.3.3 Caging Mechanism
  - 3.3.4 Charge Control System
  - 3.3.5 Vacuum Housing
  - 3.3.6 Electronics Development
  - 3.3.7 Performance Tests
- 3.4 Flight Demonstration Units
  - 3.4.1 Proof Mass
  - 3.4.2 Light-weight Proof Mass
  - 3.4.3 Reference Housing
  - 3.4.4 Charge Control System
  - 3.4.5 Vacuum Housing
  - 3.4.6 Electronics
  - 3.4.7 Functional Tests
  - 3.4.8 Integration Support
  - 3.4.9 Flight Support
- 3.5 Inertial Sensor Simulators
  - 3.5.1 Simulator Design
  - 3.5.2 Mechanical Models
  - 3.5.3 Simulated Sensor Fabrication
  - 3.5.4 Simulated Sensor Electronics
  - 3.5.5 Performance Tests
  - 3.5.6 Test-Bed Support

4 Micronewton Thrusters
- 4.1 Thruster management
- 4.2 Requirement Definition
- 4.3 Performance Tests
- 4.4 Thruster Modeling
- 4.5 Thruster Analysis
- 4.6 Lifetime Tests

5 Picometer Interferometry
- 5.1 Interferometry Management
- 5.2 Laser Subsystem
  - 5.2.1 Laser Head Acquisition
  - 5.2.2 Laser Packaging
  - 5.2.3 Control Electronics
  - 5.2.4 Performance Tests
  - 5.2.5 Environmental Tests
- 5.3 Laser Electronics
  - 5.3.1 Electronics Design
  - 5.3.2 Electronics Fabrication
  - 5.3.3 Test Equipment
  - 5.3.4 Performance Tests
- 5.4 Optical Bench
  - 5.4.1 Mounting Techniques
  - 5.4.2 Optical Modeling
  - 5.4.3 Thermal Design
  - 5.4.4 Mechanical Design
  - 5.4.5 Bench Fabrication
  - 5.4.6 Assembly Apparatus
  - 5.4.7 Bench Assembly
  - 5.4.8 Performance Tests
  - 5.4.9 Environmental Tests
- 5.5 Optical Components
  - 5.5.1 Device Selection
  - 5.5.2 Device Procurement
  - 5.5.3 Device Development
  - 5.5.4 Performance Tests
  - 5.5.5 Environmental Tests
- 5.6 Phase Measurement Electronics
  - 5.6.1 Analytic Modeling
  - 5.6.2 Electronics Procurement
  - 5.6.3 Software Development
  - 5.6.4 Performance Tests
- 5.7 Pointing Subsystem
  - 5.7.1 Device Selection
  - 5.7.2 Device Procurement
  - 5.7.3 Electronics Design
  - 5.7.4 Electronics Fabrication
  - 5.7.5 Performance Tests
  - 5.7.4 Environmental Tests
- 5.8 Rigid Interferometer Test Bed
  - 5.8.1 Interferometer Design
  - 5.8.2 Interferometer Construction
  - 5.8.3 Interferometer Operation
- 5.9 Single-Arm Test Bed
  - 5.9.1 Interferometer Design
  - 5.9.2 Mechanical Design
  - 5.9.3 Thermal Design
  - 5.9.4 Single-Arm Construction
  - 5.9.5 Single-Arm Operation
- 5.10 Three-Arm Test Bed
  - 5.10.1 Optical Design
  - 5.10.2 Mechanical Design
  - 5.10.3 Thermal Design
  - 5.10.4 Vibration Isolation System
  - 5.10.5 Vacuum System
  - 5.10.6 Lasers Procurement
  - 5.10.7 Electronics Procurement
  - 5.10.8 Test Equipment
  - 5.10.9 Software Development
  - 5.10.10 Three-Arm Operation
- 5.11 Integrated System Test Bed
  - 5.11.1 Optical Design
  - 5.11.2 Thermal Design
  - 5.11.3 Mechanical Design
  - 5.11.4 Telescope Design
  - 5.11.5 Telescope Procurement
  - 5.11.6 Instrument Construction
  - 5.11.7 Software Development
  - 5.11.8 System Operation

6 Spacecraft Simulator
- 6.1 Simulator Management
- 6.2 Interface Design/Modeling
- 6.3 Mechanical Design/Modeling
- 6.4 Thermal Design/Modeling
- 6.5 Control System Design/Modeling
- 6.5 Control Software Development
- 6.6 Simulator Software Development

4.2.3 Implementation Flow

The technology development is scheduled with an overall flow structure as indicated in Figure 4.1. The major end points of the technology development are the Flight Test of the inertial sensor, and the Integrated System Test Bed. An intermediate set of test beds is implemented to test certain subsystems and to develop the experience and infrastructure to lead to a successful Integrated System test bed.

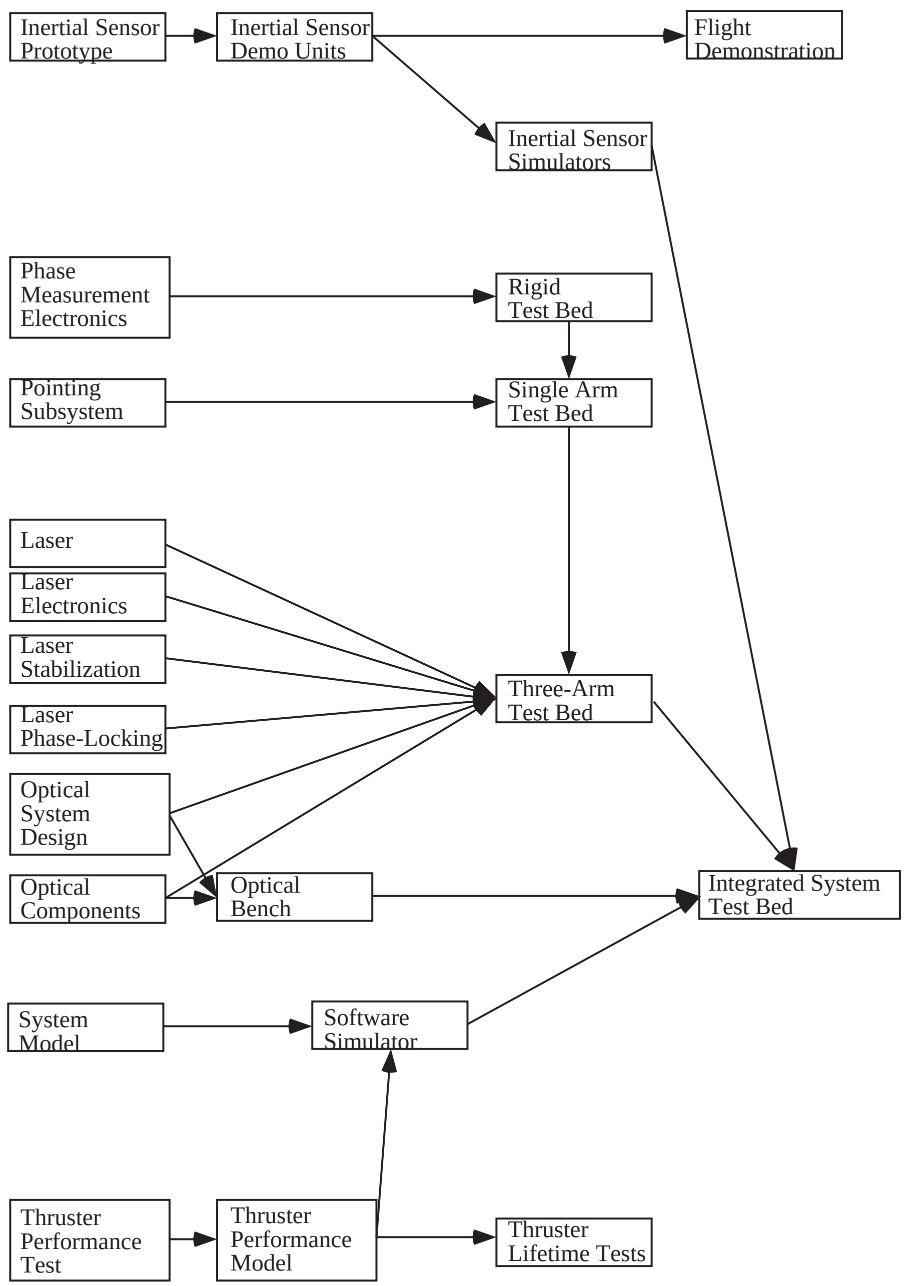

Figure 4.1 Implementation flow

4.2.4 Top Level Schedule and Milestones

The technology implementation schedule and milestones are shown in Figures 4.2 and 4.3. The tasks are ordered by the four areas; Inertial Sensors, Micronewton Thrusters, Interferometry, and Systems Operations. The inertial sensor development leading to a flight test and the sequence of interferometer test beds are the major schedule drivers.

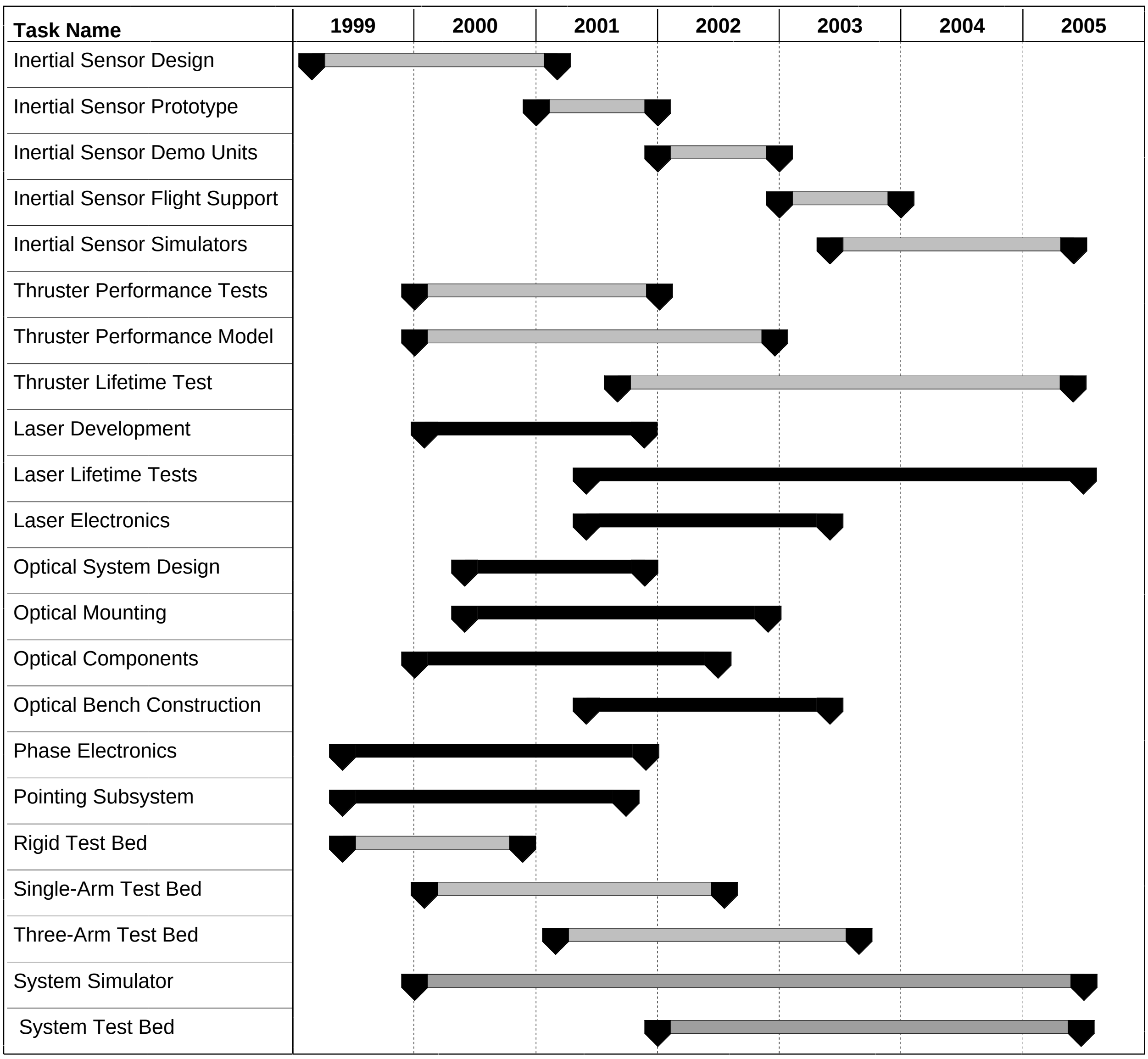

Figure 4.2 Top level implementation schedule

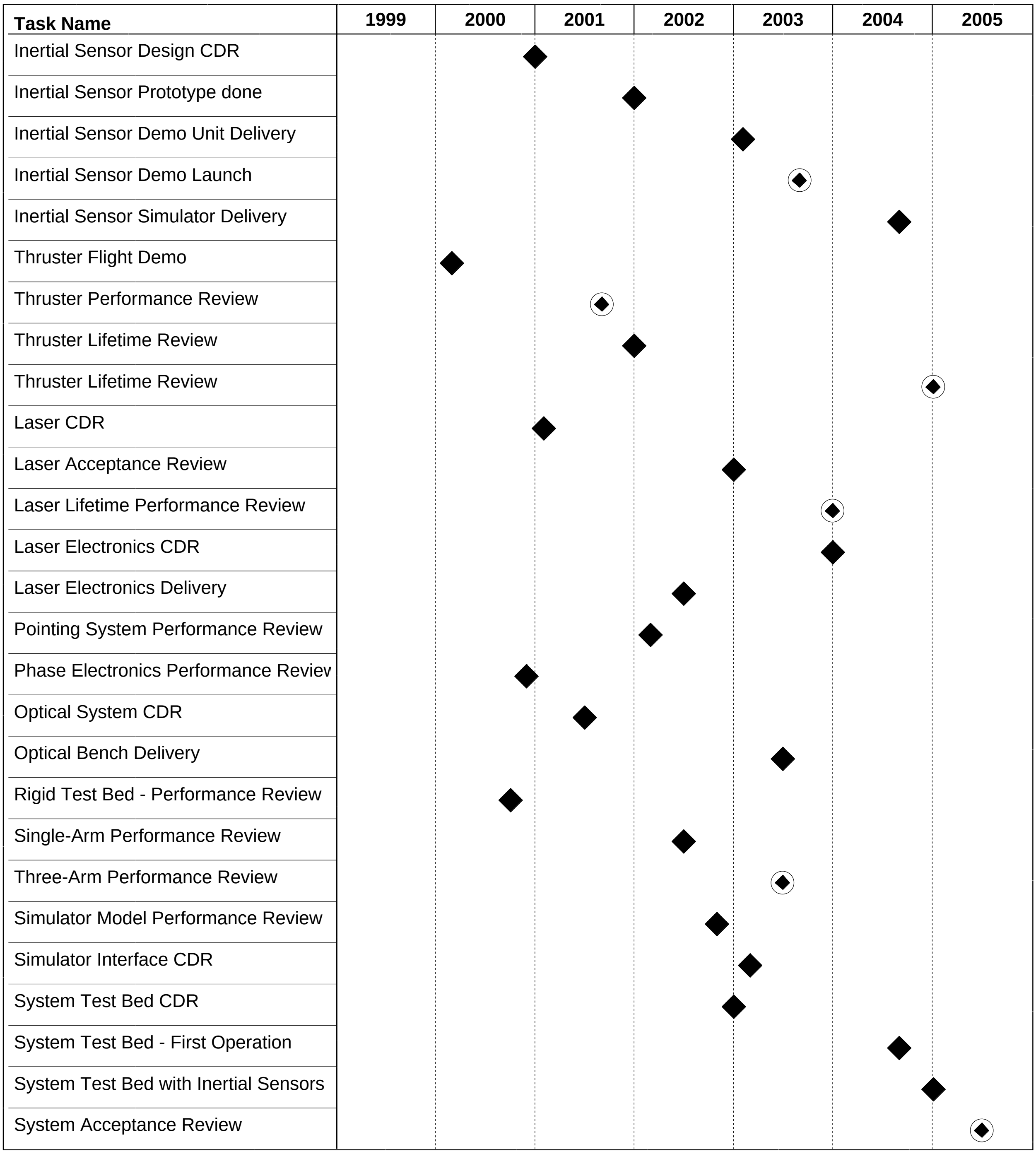

Figure 4.3 Major deliverables and milestones

4.5 Management Plan

The responsibility for the technology development described in this plan lies with the LISA project office at JPL. The project office will be responsible for coordinating efforts between NASA and ESA, and for distributing tasks between various centers of activity within the US. Separate sub-system managers are expected for the Inertial Sensors, Picometer Interferometry, Micronewton Thruster, and Spacecraft Simulator areas. Other sub-system managers will be appointed as necessary.

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A. List of Acronyms

ARISTOTELES	Applications and Research Involving Space Techniques Observing The Earth field from Low-Earth orbiting Satellite
ASTRE	Accelerometre Spatial TRiaxial Electrostatique
AU	Astronomical unit, approximately 1.5×108 km
CACTUS	Capteur ACcelerometrique Triaxial UltraSensible
CAESAR	Capacitive And Electrostatic Sensitive Accelerometer Reference
CASTOR	-- satellite name --
CDR	Critical Design Review
CHAMP	-- German geodetic satellite --
CNES	Centre National d'Etude Spatiales (France)
DSMC	Direct Simulation Monte-Carlo
ESA	European Space Agency
ESTEC	European Space Research and Technology Centre
FEEP	Field Emission Electric Propulsion
GAS	Get-Away-Special
GP-B	Gravity Probe-B
GRACE	Gravity Recovery and Climate Experiment
GRADIO	-- Gravity Gradiometry mission --
GSFC	Goddard Space Flight Center
JPL	Jet Propulsion Laboratory
LABEN	LABoratori Elettronici Nucleari
LIGO	Laser Interferometer Gravitational wave Observatory
LISA	Laser Interferometer Space Antenna
In-LMIS	Indium Liquid-Metal Ion Sources
NAR	Non-Advocate Review
NASA	National Aeronautics and Space Administration
Nd:YAG	Neodymium-doped Yttrium-Aluminum Garnet, crystal used for lasers
NPRO	Numerically PRogrammed Oscillator
ONERA	Office Nationale de'Etudes et de Recherches Aerospatiales (France)
PDR	Preliminary Design Review
PIC	Particle-In-Cell
PNAR	Preliminary Non-Advocate Review
QCM	Quartz Crystal Microbalance
RAL	Rutherford Appleton Laboratory
SILEX	Semiconductor Laser Intersatellite Link Experiment
SIM	Space Interferometer Mission
STAR	Space Three-axis Accelerometer for Research
STEP	Satellite Test of the Equivalence Principle
STSxx	Space Transportation System manifest number xx
TES	Troposphere Emission Spectrometer
TPF	Terrestrial Planet Finder
TRIAD	-- Space mission using drag free control --
UHV	Ultra High Vacuum
UK	United Kingdom
ULE	Ultra-Low Expansion
USO	Ultra-Stable Oscillator
X-band	Radio frequency for deep-space communications, near 8 GHz
WBS	Work Breakdown Structure